NASA TECHNICAL NOTE

NASA TN D-2099

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NASA TN D-2099

ENVIRONMENTAL TEST PROGRAM FOR ARIEL I

by Warner H. Hord, Jr.

Goddard Space Flight Center
Greenbelt, Maryland

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • FEBRUARY 1964

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SUMMARY

The environmental test program for Ariel I, the International Ionosphere Satellite, was accomplished at the Goddard Space Flight Center between April 27, 1961, and March 13, 1962. This program consisted of design qualification tests for the prototype spacecraft and acceptance tests for the two flight spacecraft. The spacecraft separation system and the flight vibration experiment also received design qualification and flight acceptance tests.

Several failures in the prototype spacecraft occurred during the test program and resulted in various modifications and replacements. With these changes incorporated, the two flight spacecraft completed the acceptance tests with only minor difficulties and were shipped to Cape Canaveral in mid-March 1962.

One of the flight spacecraft, Flight Unit 1, was launched into orbit on April 26, 1962. Its flight performance suggests that the test program procedures were effective and that the resulting corrective actions improved spacecraft design and operation.

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By Warner H. Hord, Jr.
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INTRODUCTION

This document presents a complete description of the environmental test program and system evaluation of the International Ionosphere Satellite, Ariel I (1962 o1), accomplished at the Goddard Space Flight Center (GSFC). Ariel I is the first of a series of three satellites to be developed as a joint effort of the United States and the United Kingdom. The experiments were designed and built by the United Kingdom, while the United States designed and built the spacecraft and supporting subsystems and provided the launch vehicle.

The purposes of Ariel I are to make a thorough 1-year analysis of the ionosphere—that is, density, temperature, x-ray intensities, and particle mass measurement—and, in general, to acquire more knowledge of the ionosphere and its relation to the sun.

Project development efforts culminated in the successful launch of the Ariel I spacecraft from Cape Canaveral, Florida, on April 26, 1962. It weighed 60.6 kg (133.6 lb), of which 17.0 kg (37.5 lb) consisted of experiments. A technical description of the spacecraft and its subsystems is contained in Appendix A.

The spacecraft was placed in elliptical orbit with an apogee of 1214 km (754 miles), a perigee of 390 km (242 miles), and a period of 100.9 min. Figure 1 shows Ariel I in an orbital configuration.

The Ariel I environmental test program was designed to produce a high degree of confidence in the ability of the spacecraft to withstand the environments expected during handling, shipment, launch, and orbital flight. The program was divided into four phases: structural model tests, functional tests (such as spinup and despin), design qualification, and flight acceptance. The *functional* tests for despin and appendage erection were conducted at 90 percent and then 110 percent of anticipated flight spin rates. The *structural model and design qualification* tests were conducted at levels considerably more severe than those expected during handling, shipment, launch, and orbital flight. In the case of mechanical tests, the environments are 1.5 times those expected from flight while, in the case of thermal tests, a temperature 10°C in excess of the expected environment is used. The *acceptance* phase consisted of tests conducted at the levels of the expected environments. These tests were conducted on the flight model spacecraft to demonstrate that the design-qualified unit had

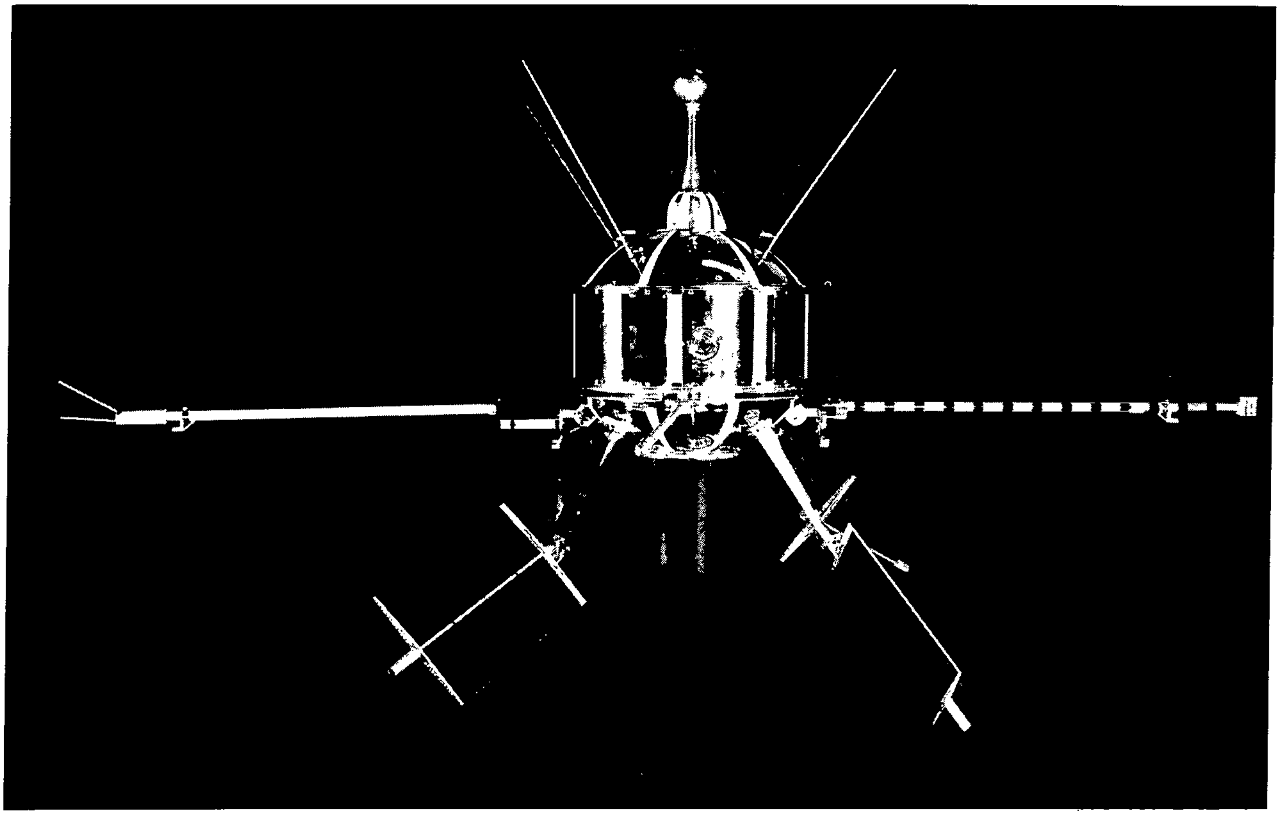

Figure 1—Ariel I in orbital configuration.

been successfully duplicated and that there were no defects in materials or workmanship. A complete test chronology of the environmental test program is given in Appendix B.

Pertinent background for the Ariel I test program is presented in the next section. The sections following cover the objectives of the test program; the proposed test plan; a description of the tests, and modifications and replacements based on test results; and, finally, an evaluation of the test results and the spacecraft system.

BACKGROUND

The original Ariel I plans called for the spacecraft to be launched by a Scout vehicle from the Wallops Island, Virginia, range in February 1962. As a result, the structural model tests were based on "Environmental Test Specification and General Test Procedures for Design Qualification and Flight Acceptance Testing of Scout Launched Satellites," August 11, 1961 (prepared by Test and Evaluation Division, GSFC); and the Scout configuration was used in the mounting arrangement. These tests were conducted between April 26 and May 9, 1961.

On September 12, 1961, NASA Headquarters assigned the Ariel I spacecraft to a Delta launch vehicle with launch to be from Cape Canaveral in March 1962. It was determined unnecessary to repeat the structural model tests because they had been successful for Scout, and test levels for a Delta launch would not be significantly different. The change in launch vehicle did require that an adapter section, approximately 30.5 cm (12 in.) high, be inserted between the final-stage rocket motor and the spacecraft separation system; this adapter is referred to throughout this report as the Dutchman. Furthermore, the responsibility for procuring and testing the spacecraft separation system was transferred from Langley Research Center to GSFC.

Because of the added space provided by the Dutchman and the increase in spacecraft weight permitted by the greater thrust of the Delta, a separate telemetry system that made possible an experiment for measuring vibration during the period from launch to the separation of the spacecraft from the third stage was installed in the Dutchman. This telemetry system had three subcarrier oscillators for vibrational data and a fourth subcarrier oscillator that provided data on nose-fairing contamination, closure of the third-stage pressure switch, spin rate, and aspect angle. Appendix C contains a description of the vibration telemetry system. As a result of these changes, tests of the separation system and vibration experiment had to be included in the environmental test program.

The design qualification test program of the prototype Ariel I spacecraft was conducted during the period September 16, 1961, through January 8, 1962. Because of modifications and changes determined necessary during the qualification tests, the vibration and thermal-vacuum tests were repeated during the period February 12 through March 12, 1962. Design qualification tests of the vibration experiment and separation system were conducted during the period January 15 through January 18, 1962, and were repeated during the period March 10 through March 15, 1962.

Acceptance tests of Flight Unit 1 were conducted during the period January 11 through February 27, 1962; and Flight Unit 2 acceptance tests were conducted February 5 through March 13, 1962. Acceptance tests of the flight separation system and vibration experiment were conducted during the period January through March 1962.

The Prototype Unit was shipped to the Atlantic Missile Range (AMR) on March 13 for coordination and interference checks with the launch vehicle. Subsequently, Flight Units 1 and 2 were shipped.

After checkout, Flight Unit 1 was selected for launch. Operations at AMR were relatively smooth with the only significant problems being that (1) the electron density experiment boom was damaged, requiring substitution of the Flight Unit 2 boom; and (2) a gear shaft of the boom escapement mechanism failed, and required replacement. (A chronology of AMR operations is given in Appendix D.) The first launch attempt on April 10, 1962, was cancelled because of problems with the second stage. A 2-week slip was necessary so that the second stage of the booster could be replaced. The second launch attempt on April 26, 1962, was successful; and at 1300 hours EST, after a hold of 1 hour, the Ariel I Flight Unit 1 was launched into orbit.

TEST OBJECTIVES

General

The four test programs in which the Ariel I project was involved—structural model, functional, design qualification, and flight acceptance—had distinct purposes as described below. These programs had the common purpose of providing a spacecraft of maximum reliability at a reasonable cost by using test levels and procedures reasonably related to predicted environments of the Ariel I spacecraft.

Further, the series of environmental exposures provided an unparalleled opportunity for the operating crew to gain experience with the spacecraft system prior to operations at the launch site—thereby serving as an excellent training program.

Structural Model

The purpose of this program was to subject a structural model of the spacecraft with dummy components to environmental exposures up to the design qualification level in order to "prove out" the structural design prior to building a prototype spacecraft with operating subsystems. Thus, considerable confidence could be established prior to design qualification tests that the prototype spacecraft structure would withstand required levels of exposure.

Also, data from these tests were needed to establish levels in test specifications for subsystems to be mounted in the spacecraft.

Functional

This test program had the objective of determining the ability of a dynamic mockup of the spacecraft to perform functional requirements such as spinup, despin, boom erection, paddle deployment, and separation from the third stage.

Design Qualification

The design qualification tests for the prototype spacecraft system and subsystems had the purpose of establishing a high degree of confidence that the spacecraft system and subsystems as designed would not be impaired by predicted environments, and thus would be suitable for required operations. Since this confidence had to be established by testing just one prototype, normal statistical sampling was not possible. Instead, test specimens were subjected to considerably greater rigors of environment than expected during ground-handling, launch, and flight.

Flight Acceptance

The acceptance tests for the flight spacecraft system and subsystems were designed to locate latent defects in material and workmanship, thereby providing assurance that none of the essential characteristics of design had been degraded during manufacturing and the accompanying inspection and handling. An additional objective was to demonstrate the compatibility of the subsystems and other elements of the spacecraft system under simulated launch and orbital environments. The levels of the flight acceptance tests approximated predicted environmental conditions. This choice of test levels is based on the philosophy that this degree of exposure allows detection of latent defects in material and workmanship without the unnecessary risk of damage from exposure above predicted levels. In other words, the spacecraft is tested through its early operating lifetime or infancy when its parts mortality is greatest according to past experience.

TEST PLANS

General

The Ariel I environmental test program was mainly accomplished at GSFC. A description and evaluation of the test results constitute the basic purposes of this report.

The test program consisted of mechanical tests at the design qualification level applied to the structural model of the spacecraft; design qualification tests for one spacecraft (the Prototype) as well as prototype models of the vibration experiment, separation system, and one prototype solar paddle; and flight acceptance tests for two spacecraft (Flight Units 1 and 2), the vibration experiment, and spacecraft separation systems.

Structural Model

Engineering Test Unit 1 (ETU 1), a model of the Ariel I spacecraft structure, was to be statically balanced and subjected to exposures of spin, acceleration, and vibration at design qualification levels to determine the balance and response of ETU 1 to these exposures.

ETU 1 was to contain no electronic equipment, but the intended flight assemblies were to be simulated in respect to weight and size by dummy weights placed in appropriate positions in the model.

Functional

Inertially correct mockups, including all essential hardware, of the Ariel I spacecraft and final-stage X-248 booster were to be sequentially subjected to spinup by PET rockets, yo-yo despin, experiment boom erection, solar paddle and mass boom deployment, and separation of spacecraft from the X-248.

Design Qualification

Balance

Static and dynamic balancing of the spacecraft is necessary to insure spin stability of the spacecraft during launch and orbital flight. It was specified for the prototype spacecraft so that this unit would be dynamically similar to the flight unit during acceleration and vibration testing. Balancing of the prototype was mainly for determining suitable locations for balance weights as well as determining that methods planned for balance of the flight units were suitable.

Spin

The prototype spacecraft was to be spun at 225 rpm for 1 min and at 150 rpm for 30 min to exceed the predicted 150 rpm experienced by the spacecraft third-stage combination from third-stage ignition. Operation of spacecraft was to be checked during spin.

Acceleration Test

The maximum acceleration (18g), imparted to the Ariel I spacecraft by the Delta launch vehicle, occurs just prior to third-stage burnout. The orientation of the spacecraft on the centrifuge was selected so as to simulate the sustained loading of this thrust-induced acceleration. In addition, transverse acceleration tests were specified based on expected handling loads of 2g. Figures 2 and 3 show the prototype before and after installation on the centrifuge.

Shock Test

A shock environment is produced in several ways—handling, shipment, stage ignition, and stage separation being the most common. The Ariel I shock test parameters were dictated by handling and transportation considerations, since the shock pulses generated by the Delta launch vehicle were expected to be less severe.

Temperature and Humidity Tests

This series of tests had the purpose of simulating the temperature and humidity conditions of storage, transportation, and pre-launch periods to determine the effect of those conditions on the Ariel I spacecraft. The operational temperature tests are based on expected flight temperature and are conducted prior to thermal-vacuum tests.

The program was planned in five parts: (1) -30°C soak, (2) +60°C soak, (3) -10°C operational, (4) +50°C operational, and (5) humidity exposure. Parts 1 and 2 were to consist of 6-hour periods of exposure at each temperature with performance checks at ambient temperature (25°C) after each exposure. Parts 3 and 4 were to consist of monitoring spacecraft performance at the stabilized temperature of -10°C and +50°C. Performance checks also were to be performed at ambient temperature after each exposure. Part 5 was to consist of a 24-hour exposure to 30°C and a relative humidity of 95 percent. At the end of the exposure period, spacecraft performance was to be checked.

Figure 2—Prototype setup prior to installation for acceleration.

Figure 3—Prototype spacecraft installed on centrifuge for acceleration.

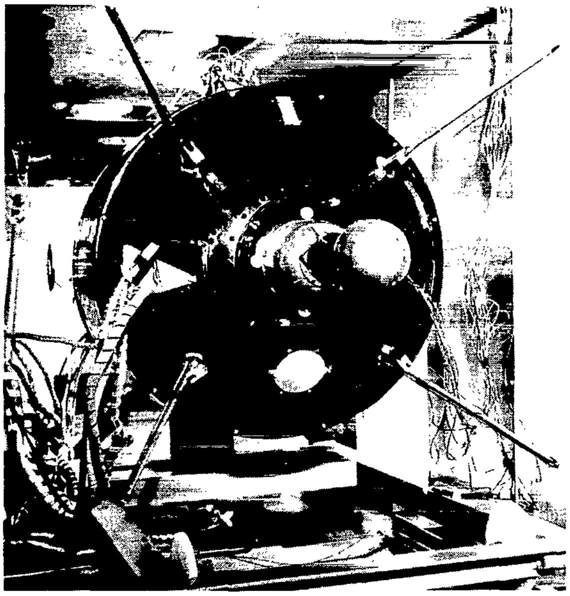

Figure 4—Setup for prototype temperature and humidity test.

Spacecraft performance was to be monitored during the drying-out period. Figure 4 shows the setup for these tests.

Vibration

Vibratory excitation of the spacecraft arises from shipment and rocket motor burning, as well as from acoustic and aerodynamic sources. Random excitation and combustion resonance tests along the thrust axis and two lateral axes were to be conducted to simulate conditions anticipated in a launch by the Thor-Delta vehicle.

Vibration testing of the Ariel I spacecraft was complicated by the fact that there are several appendages (electron density boom, solar paddles, etc.) which, prior to separation, are folded down along the case of the last-stage X-248 rocket motor.

To simulate this situation, it was necessary to fabricate a cylindrical structure of the same nominal diameter as the rocket motor and of adequate length to accommodate the appendages, yet retaining the desirable stiffness properties of a good vibration fixture. The ideal vibration fixture is a massless, infinitely stiff device designed to adapt the test specimen to the shaker table. It was recognized that building such a fixture with no resonances within the applicable frequency band would be impossible. The approach taken was to build the fixture as stiff as practical, and then to check it out in vibration with a dummy load simulating the Ariel I system. When this was done, it was found that the setup was usable to about 1000 cps in the thrust direction and about 200 cps in the lateral direction.

Because of the frequency limitations imposed by this setup, each of the tests was divided into two parts. The lower frequency tests were to be performed with the complete system, including the appendages, in place of the cylindrical fixture (which was called the "tall fixture"). Higher frequency tests were to be performed without the tall fixture and without appendages, using a simple flat plate (short fixture) to adapt the shaker table to the Dutchman interface. Figures 5 and 6 present the tall and short fixtures, respectively. It will be noted that in following this procedure the various booms and paddles were not to be tested above 1000 cps in the thrust axis or 200 cps in the lateral axes.

Additional subsystem tests of these appendages were planned, but were waived on the basis that the appendages were essentially decoupled from the vibration input at the high frequencies.

Thermal-Vacuum Tests

Background: These tests were to be conducted to simulate the extremes of temperature in a vacuum environment that the spacecraft was expected to encounter in orbit. Satisfactory

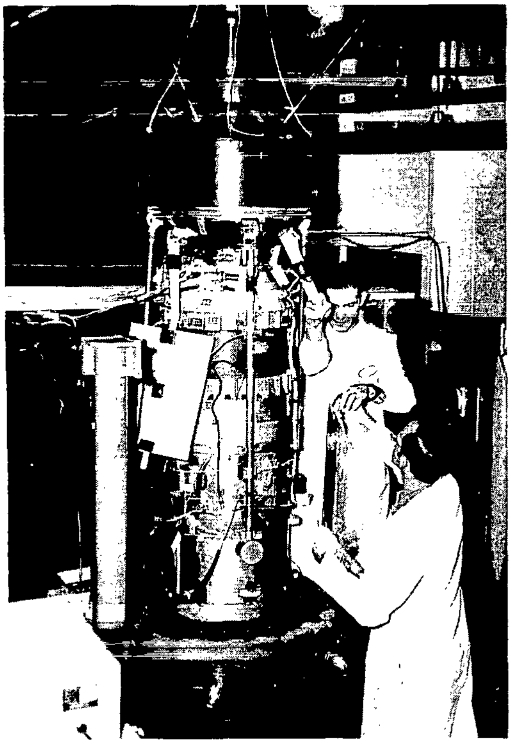

Figure 5—Installing solar paddle prior to thrust axis test of prototype unit on tall fixture.

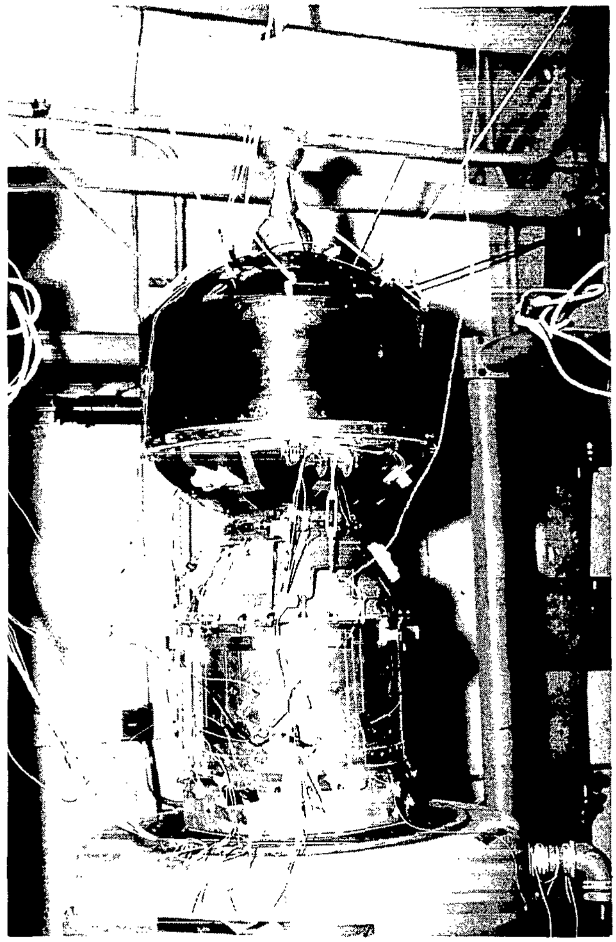

Figure 6—Thrust axis setup using short fixture.

operation of the spacecraft system at -8°C and +35°C with a vacuum of 1×10^{-5}mm Hg was to be verified.

During its orbital life of 1 year, the spacecraft will have several aspect positions relative to the sun. This means that heating of one side occurs while the opposite side is being cooled; and, later in orbital life, the heated and cooled surfaces will be reversed. The test procedure was to cover the simultaneous heating-cooling type of environment as well as the conventional uniform hot or cold environment.

Description: The test was to consist of four parts: (1) cold, (2) hot, (3) 30° aspect, and (4) 135° aspect. Test 3 was to simulate the space environment where the sun is at an angle of 30 degrees with respect to the longitudinal spin axis of the spacecraft and illuminates the top portion of the system. Test 4 was to simulate the space environment where the sun is at an angle of 135 degrees with respect to the longitudinal axis of the spacecraft illuminating the bottom portion of the system. (The projected area is a maximum at the 30° aspect and a minimum at 135°.) Tests 1 and 2 were to be conventional tests in which the entire spacecraft was to be soaked at a uniform temperature.

The external power supply was to be such that:

1. Spacecraft could be operated solely on external power.
2. Spacecraft could be operated solely on internal power.
3. Batteries could be charged during the test using the shunt regulator.

Flight Acceptance

Balance

Both static and dynamic balancing of the flight spacecraft were specified to insure spin stability of the spacecraft during launch and orbital flight. Figure 7 shows the Ariel I spacecraft on the balance machine.

Spin

The flight spacecraft were to be spun at 150 rpm to verify the spacecraft system operation at the rate of spin expected prior to third-stage ignition.

Vibration

Vibration levels no greater than those expected during launch and injection into orbit were to be applied in each of three orthogonal directions.

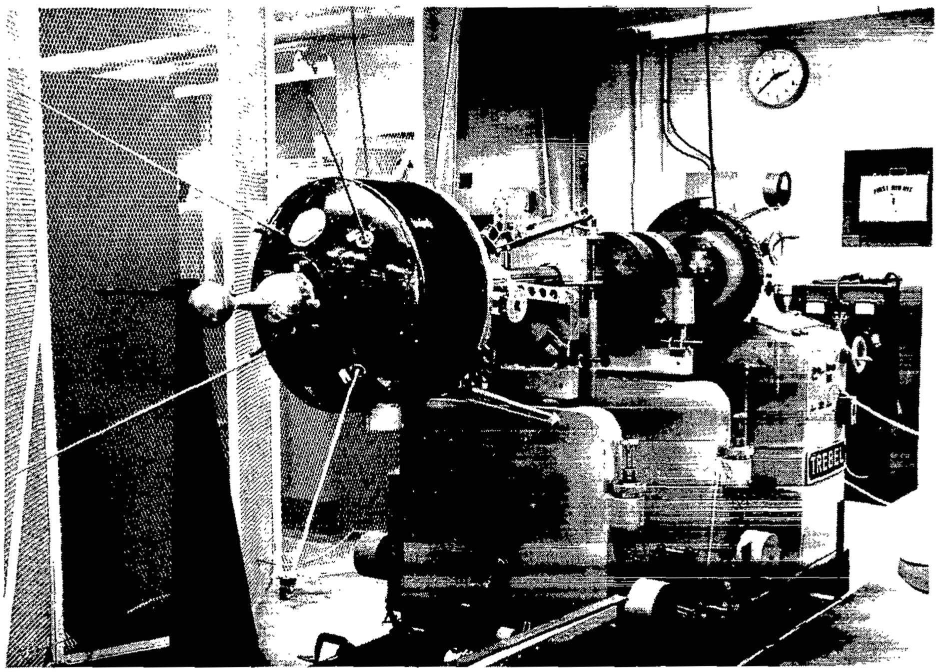

Figure 7—Ariel I balance configuration.

The flight spacecraft were to be operated in a duty cycle typical of that to be employed in actual launch and monitored for malfunctions in telemetering and other systems that operate during boost.

Thermal-Vacuum

Flight spacecraft were to be exposed to predicted vacuum and temperature. Predicted solar conditions were to be simulated. The operation of the flight spacecraft was to be monitored during above exposures.

TEST RESULTS

Structural Model

The structural model of the spacecraft was tested from April 7 through May 9, 1961. The test environments included static balance, spin, acceleration, and vibration. These tests were conducted at increasing levels of severity up to design qualification levels. Vibration equipment consisted of two 2250 kg (5000 lb) exciters; one was used for thrust axis excitation, and the other for lateral axis excitation.

The static unbalance of this unit was reduced from 1.5×10^7 dyne-cm (218 oz-in.) to 4.4×10^6 dyne-cm (61.8 oz-in.) by the addition of 0.68 kg (1.5 lb) of lead to the main shelf.

The main difficulties arose during vibration. During vibration in thrust axis at acceptance level, several screws holding the support tube to the main shelf sheared; this problem was solved by increasing the size and number of screws. The second failure of screws occurred during lateral vibration at acceptance level when several that fastened the struts to the main shelf failed; this was tentatively solved by increasing the number and size of screws at this location and assuring that the countersink for the screw heads was concentric with the through-hole. The third failure of screws occurred during vibration in the thrust axis of design qualification levels; in this case the screws fastening the struts to the support tube failed. The solution adopted was to add shear pins between the support tube and struts.

The design of subsequent units was changed to include a machined ring on the support tube and slots on the struts which, when assembled, would act as a shear pin. In addition, shear pins were used at the joint between the struts and the main shelf.

The structure successfully passed the design qualification level exposures after the above-mentioned modifications.

Response data for use in subsystem testing were taken during these vibration tests. However, because of the change from the Scout to the Delta launch vehicle—requiring the addition of an adapter section (the Dutchman), the data were no longer valid.

Functional

The functional tests were conducted on inertially correct mockups of the Ariel I spacecraft and final-stage X-248 booster between October 16 and November 16, 1961. All tests were run under vacuum conditions in the 18.3 m (60 ft) sphere at Langley Research Center (LRC).

Results of the functional test program were generally satisfactory, and details are presented in Appendix E.

Prototype

Balance

Initial balance operations resulted in addition of a 0.454 kg (1 lb) balance weight, bringing the total weight of the spacecraft to 48.4 kg (106.6 lb). Because of difficulty in removing protective devices for spacecraft experiment sensors, balancing was conducted without their removal; but the method of attachment was changed so that covers could be removed for later balancing operations.

Spin

The spacecraft was spun at 225 rpm for 1 min and at 150 rpm for 30 min. During initial turn-on operations, the spacecraft went into undervoltage on insertion of the turn-on plug because of low battery voltage. This condition was corrected, and a satisfactory test was conducted.

Temperature and Humidity

General: The operational temperature test revealed problems in the electron density experiment, tape recorder, and cosmic ray experiment. However, it was decided not to repeat this exposure, since an operational temperature test under vacuum was to be conducted in the thermal-vacuum test program to follow. Furthermore, required replacement parts for the tape recorder and cosmic ray experiment could not be procured in a short time. The x-ray and mass spectrometer experiments showed some effect from high humidity.

Cosmic Ray Experiment: During the systems checkout and on completion of the +60°C exposure test, excessive counts were observed for the cosmic ray experiment. As the temperature test progressed, degradation continued. This condition was due to a defective Geiger tube. The malfunction could not be corrected because replacement tubes were not available at that time, but the faulty tube was later replaced.

X-Ray Experiment: The x-ray experiment showed some effect on completion of the 24-hr exposure at 30°C and 95 percent relative humidity: The experiment was drawing excessive current. However, during the first 5 minutes of the drying period (RH = 80 percent), the current returned to normal and its operation continued satisfactorily.

Mass Spectrometer Experiment: The effect of humidity was also observed in the output of the mass spectrometer experiment. Four test points were below minimum tolerance. However, during the first 10 minutes of the drying period (RH = 42 percent), these readings returned to within the minimum tolerance.

Electron Density Experiment: During the initial -10°C operational test, improper operation of the electron density experiment was observed: The experiment was found to be gating at random. The output indicated maximum (6 to 7 volts) and could not be varied when the capacitance of the sensor was varied. At that time it was believed that the malfunction was a result of a temperature gradient within the electronics (this being due to the short stabilization period of 2 hr). Two additional thermocouples were installed within the electronics pack on completion of the initial test. During the final -10°C test, the experiment again failed to operate at low temperature. Operation was checked at 5-degree intervals from +5°C to -10°C. Improper operation appeared at +5°C and continued as the temperature was reduced to -10°C; no further attempt was made to correct the condition at that time. Operation was satisfactory at +50°C and room temperature. Since the experiment was not designed to operate under 0°C and the experimenter did not wish to redesign it, thermal coatings designed to prevent temperatures below 0°C were developed for the boom electronics can.

Tape Recorder: Also, during the initial -10°C test it was observed that the tape recorder was drawing excessive current and was operating at a slower than normal speed. This condition was also experienced during the final low temperature exposure; at that time, improper operation was observed at -5°C and continued as the temperature was reduced to -10°C. No further action was taken to correct the condition at that time. Operation was satisfactory at +50°C and at room temperature.

The prototype tape recorder—which would not operate satisfactorily below -5°C—was replaced subsequent to this test by the tape recorder planned for Flight Unit 1, which had been modified since delivery of the prototype recorder. The later unit had been operated by the designer satisfactorily at -20°C.

Vibration

General: For the first prototype test, the spacecraft was installed atop a fixture that was hinged at the floor and driven at the fixture—Dutchman interface. By so doing, the input was angular motion rather than the desired true translational input. Consequently, components near the top of the spacecraft would receive a somewhat higher input than desired.

For the prototype retest and flight unit tests, all thrust axis tests were done on the 4500 kg (10,000 lb) vibration exciter. The lateral axes tests were performed on a hydrostatic bearing table driven by the same shaker. The hydrostatic bearing table represents an innovation in vibration testing—being essentially similar to the slippery tables or oil-film tables commonly used for this purpose, but much more effective in constraining lateral and vertical cross axis motion, as well as being able to withstand much higher bending moments without deleterious effects. The result is the closest approach to true unilateral motion believed to be attainable thus far.

Because of the large mass of the system it was not possible to get the required level of ±86g for the prototype combustion resonance test with the shaker. Therefore, the shaker was driven to its maximum output, which was about ±60g—slightly above the flight level. This represented the only deviation from the specified test program.

Although no structural failures occurred in the initial series of tests, conducted in October 1961, there were several discrepancies that were attributed in part to the severity of the fundamental thrust axis resonance.

Post Test 1: To reduce the severity of the thrust axis resonance, the Dutchman (extension section) was redesigned to provide greater stiffness and damping. All tests of the second series, conducted in February 1962, were completed satisfactorily with one exception: When main thrust axis resonance (85 cps) was reached, the yo-yo despin weights shifted considerably, shearing some of the lead from the weights. As a result, the despin weights on all units were changed from lead to brass.

Cosmic Ray Experiment:

Test 1: The photomultiplier tube failed.

Post Test 1: After exhaustive testing, the photomultiplier tube was replaced by a redesigned unit.

Electron Density Experiment:

Test 1: Operation was intermittent.

Post Test 1: The electron density grids were found to have been previously damaged and were replaced.

Thermal-Vacuum

General: For this test program, a 2.4 × 2.4 m (8 × 8 ft) thermal-vacuum chamber was used. It has a temperature range from -65° C to +100° C and can achieve an ultimate vacuum of 6×10^{-8} mm Hg.

The spacecraft was mounted on the test fixture, in the thermal-vacuum chamber, with the spin axis horizontal. Three solar paddle arms were attached (paddles not available) and extended, and the fourth arm was removed. The electron temperature and the electron density booms were attached to the structure in the folded position. The spacecraft was supported by a test fixture attached to one of the spacecraft struts and to the forward end of the spacecraft mid-skin structure. The protective strip coat was removed from the structure after completion of all instrumentation. Figure 8 indicates the method of mounting and the location of the two heat lamp rings that were utilized during the 30 degree and 135 degree solar aspect tests. Figure 9 shows the general arrangement of the spacecraft being prepared for thermal-vacuum testing.

Numerous difficulties extended this test program from a normal 2 1/2 weeks to 9 weeks and caused the spacecraft to experience three cycles each of low and high temperature instead of the planned program of one test of each type. Figure 10 shows the spacecraft checkout equipment used.

Major problems required the redesign of the x-ray and cosmic ray experiments, the tape recorder, and the shunt regulator. In addition, it was determined that the accuracy of the electron density experiment was reduced at low temperatures and that the U.K. converter required heat sinks. Also, there were a number of component problems in the spacecraft circuitry that were routinely resolved.

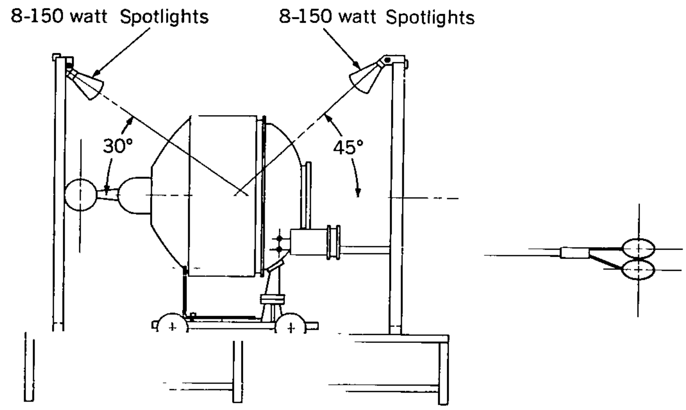

Figure 8—Thermal-vacuum test setup.

X-Ray Experiment:

Test 1: During initial chamber evacuation, the high voltage developed by

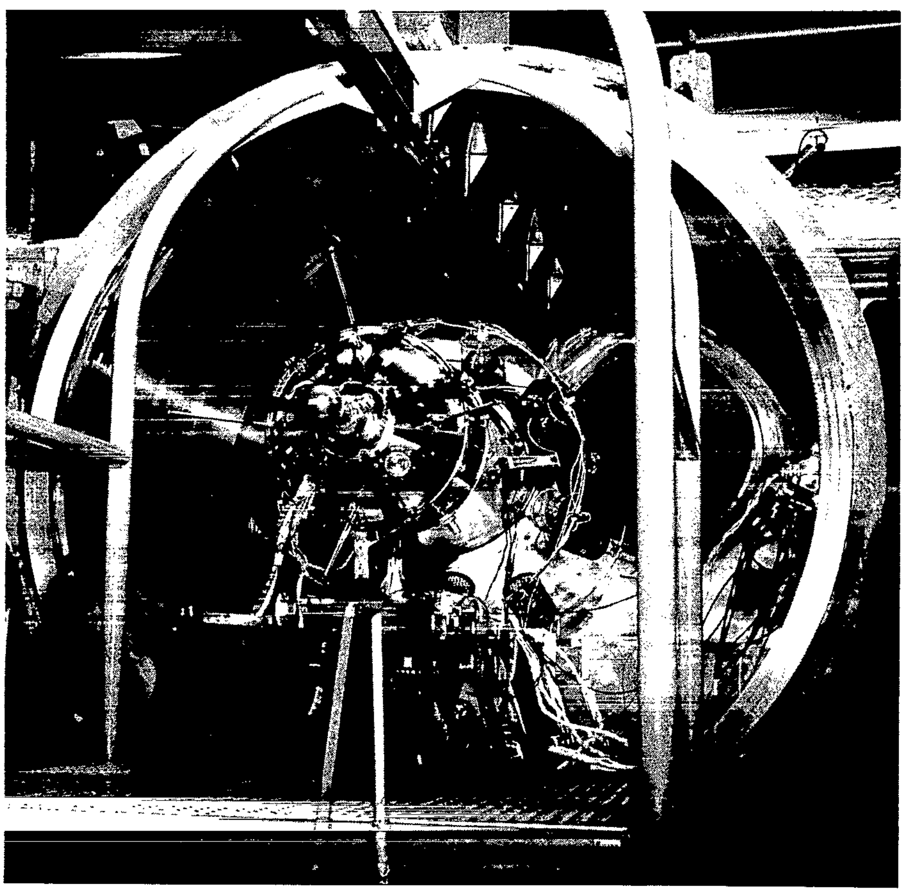

Figure 9—Ariel I prototype system prepared for thermal-vacuum testing.

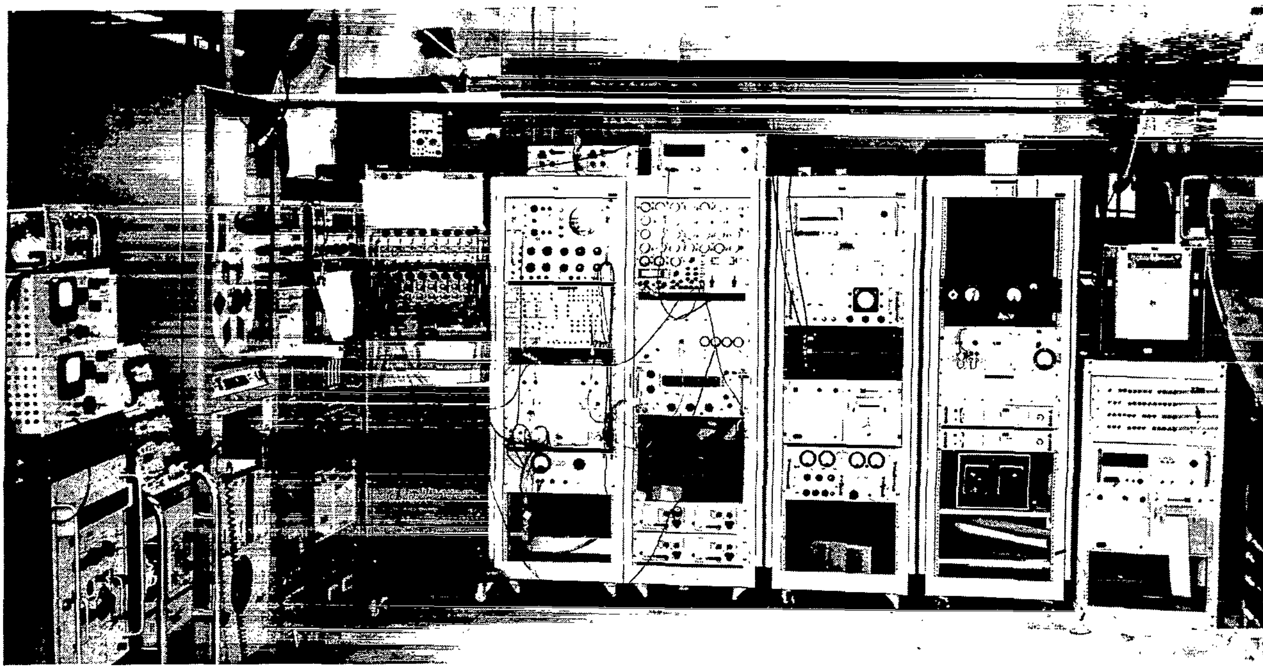

Figure 10—Spacecraft checkout equipment used during thermal-vacuum testing.

the x-ray extra high tension (EHT) card failed at 70 mm Hg. Additional insulation applied to the high voltage lead connectors and sensors did not resolve the difficulty. Prior to the third cold test, the EHT card was replaced by a dummy.

Post Test 1: A transformer in the EHT card was replaced by a vacuum-impregnated high voltage unit; and one capacitor, operating above its rated voltage, was replaced with a new type.

Test 2: The x-ray experiment was successfully operated through the corona region. Arcing of the high voltage occurred at approximately 87 mm Hg, but operation returned to normal at approximately 5×10^{-2} mm Hg (50 microns).

Cosmic Ray Experiment:

Test 1: A large Geiger tube failed during the +55°C vacuum test. Since a suitable replacement could not be obtained in a reasonable length of time, this tube was permanently removed from the experiment. During the -10°C vacuum test one binary, in a chain of 15, was observed to be sensitive to low temperatures, resulting in excessive counts. Because of its location within the circuit no attempt was made to correct the problem; it did not recur on the flight units in the acceptance test program.

Electron Density Experiment:

Test 1: During the 30 degree solar aspect test, this experiment operated improperly—there being a single period of 138 μsec in place of the normal two sawtooth pulses. This was caused by the low temperature to which the boom was exposed for this test.

Post Test 1: A thermal coating of evaporated gold and black paint was applied to the boom. This coating was designed to maintain the temperature of the boom electronics between 11° and 51°C during orbital flight, and thus avoid low temperature problems in this experiment.

Tape Recorder:

Test 1: The tape recorder failed during the first -10° C test because of a broken magnetic tape splice, shunted ground wire, and failure to pre-load motor bearings. Later, on return to -10°C, the tape recorder did not turn off when system was in undervoltage condition and drew excessive current during the +47°C test.

Post Test 1: The broken tape splice was mended, the short repaired, and the bearing pre-load adjusted. In addition a fail-safe power control circuit was designed to turn the tape recorder off when current exceeded 150 ma for 70 sec. This circuit also was to respond to the playback command to the tape recorder by switching power on to tape recorder. If the current continued to be excessive, it would again turn the tape recorder off.

Test 2: During the low temperature test, the tape recorder began to draw excessive current and was turned off by the fail-safe circuit installed after Test 1.

Post Test 2: Investigation revealed that a rib within the housing was causing the tape recorder to bind and thus draw excessive current. The housing was modified.

Test 3: The tape recorder operated satisfactorily after corrections of the following mechanical problems: loose drive belt, misaligned drive shaft, and broken drive belt.

U.K. Converter:

Test 1: A high-temperature condition occurred during the initial +55°C and was corrected by addition of a suitable heat-flow path from the card to the instrumentation shelf.

Shunt Regulator:

Test 1: Redesign was accomplished to electrically isolate the regulator from the structure when it was discovered in initial setup that the regulator was grounded to the structure. The regulator was also redesigned to prevent batteries from being charged at an excessive rate after it was found during the +47°C test that the regulator had no current-limiting capability, resulting in a high-temperature condition (+72°C) in the batteries.

Post Test 1: Redesign resulted in the limitation of current and voltage at +70°C to 0.3 to 0.4 amp with 13.6 volts and at -10° C to 1.2 amp at 14.2 volts. Also, the battery pack was modified to improve the surface contact between it and the instrumentation shelf. These modifications were to reduce the temperature of the batteries to a tolerable level.

Special Test Programs

Solar Paddle: The prototype solar paddle, serial no. 1, was subjected to the following environments:

1. Temperature storage, -30°C and +60°C, 6 hr each
2. Humidity, 95 percent relative humidity at 30°C for 24 hr
3. Vacuum, 5×10^{-4} mm Hg, ambient temperature
4. Temperature shock, from +26°C to -65°C in 20 min, to +50°C in 1 hr, to +26°C in 20 min; this cycle was repeated three times
5. Temperature shock, from +26° to 100°C, back to 26°C
6. Thermal-vacuum, -10°C and +55°C at 1×10^{-5} mm Hg

The paddle was checked for insulation resistance and diode leakage, and visually checked for changes in the surface. No problems were encountered.

Vibration Experiment:

Temperature and Humidity: The low- and high-temperature storage consisted of 6-hr exposures at -30°C and +60°C, with a complete checkout of the system before and after each exposure. The humidity exposure consisted of 24 hr at 30°C and 95 percent relative humidity; the system was checked out before and after the test; no apparent humidity effects were observed. During the low-temperature test (0°C), it was found that the external power supply had to be reduced from 28 to 26 volts for the system to calibrate properly. Operation was normal during the high-temperature test, which was performed at +40°C.

Vibration: The simulated battery box separated from its mounting brackets because the six spot welds were not strong enough. Subsequently, the angle brackets supporting the battery pack for the vibration experiment were improved by changing from spot welding to continuous welding.

Thermal-Vacuum: Operation of the system was normal during both the low- and high-temperature thermal-vacuum tests. The system was monitored through the corona region and showed no adverse effects. The low-temperature thermal-vacuum test was conducted at 0°C with the system operated on internal power for 15 min. The high-temperature vacuum test was conducted at +50°C. The system was again operated for 15 min on internal power.

Separation System: This system was mounted in the normal configuration with the spacecraft and Dutchman for the second series of prototype spacecraft vibration tests. For the remainder of the second round of the prototype environmental test program, the separation system was separately tested.

No problems arose from the prototype test program except that, during vibration, the operation of the mechanical timers in two redundant subsystems was not satisfactory. No pair could be selected from the eight available that could match each other's time within the required 2 percent in performing the necessary operational sequence. Therefore, electronic timers were designed and fabricated in time for the flight acceptance environmental test program.

Flight Unit 1

Balance

Balance operations added 0.844 kg (1.86 lb), bringing the total weight of the spacecraft to 49.56 kg (109.26 lb).

Spin

The spacecraft was spun at 150 rpm for 30 min. During this time unbalance readings were taken, and telemetry was transmitted and recorded on tape.

Vibration

No problems were encountered.

Thermal-Vacuum

Irregularities were noted, and action was taken as described below.

Cosmic Ray Experiment: This experiment would not turn on at -8°C; but operation became normal at +10°C, and it would operate at -8°C if the temperature were reduced with the experiment in operation. No action was taken.

X-Ray Experiment: The experiment was not gating properly because of a faulty diode that subsequently was replaced.

Electron Energy Experiments 1 and 2: Experiment no. 1 malfunctioned because of an open capacitor that was replaced. Memory cards for both experiments were modified to eliminate the problem of digitizing low-speed data.

Shunt Regulator: Transistors operated at excessive temperatures because of an inadequate heat sink. The method of mounting was modified to improve the conductive path to the solar paddle arms.

Electron Density Experiment: The operation of this experiment was questionable during the hot test, so a modified waveform generator card was installed.

Special Test Programs

Vibration Experiment: No problems were encountered.

Separation Assembly: There were no problems except that, during vibration, the two mechanical timers required replacement by GSFC electronic timers, developed as a result of the prototype testing.

Flight Unit 2

Balance

Balancing operations added 0.69 kg (1.52 lb) to bring the final spacecraft weight to 49.4 kg (109 lb).

Spin

The spacecraft was spun for 30 min at 150 rpm. Operation was satisfactory.

Vibration

Cosmic Ray Experiment: There was a malfunction during the thrust axis random test because of failure in weld of the photomultiplier tube. To correct this situation, a spare cosmic ray unit was qualified and installed as a replacement for the defective unit.

Antenna: The antenna lost power because of a loosened micro-dot connector, which was tightened.

Mass Spectrometer: Faulty operation of this unit disclosed that one of the sphere connector pins had sheared, apparently because the mounting nut had not been sufficiently tightened.

Thermal-Vacuum

General: The following actions were taken prior to thermal-vacuum tests as a result of the behavior of Flight Unit 1 and the Prototype Unit in thermal-vacuum exposures:

1. The method of mounting the shunt regulator power transistors was modified to improve the conductive path to the solar paddle arms.
2. A modified waveform generator card was installed in the electron density experiment.
3. The tape recorder housing was modified, and the terminals were insulated.

Mass Spectrometer: This equipment malfunctioned because of a broken pin in the sphere. The sphere was replaced.

Tape Recorder: Operation of the dc control unit was questionable in checkout because of a non-qualified circuit design. The problem was resolved by installing a unit modified in accordance with prototype test results.

The tape recorder malfunctioned in the cold test because of a faulty idler bearing, which was replaced. Malfunction reoccurred after 10 hrs of hot test. After tests, the idler bearing and drive motor were replaced.

Electron Density Experiment: This experiment malfunctioned, but the malfunction could not be reproduced on a subsystem basis. Voltage spikes on several supply lines were determined to be the cause. (This problem did not occur in Flight Unit 1, which was flown.)

Special Test Programs

Solar Paddles: No problems were encountered.

Vibration Experiment: There was no Flight Unit 2 vibration experiment; no backup unit was available.

Separation System: It was again verified during vibration that the mechanical timers could not meet operational requirements and thus installation of the GSFC electronic timers was necessary. This was the only problem.

SUMMARY AND EVALUATION

Structural Model

The structural model test program revealed one major weakness in the structural design: Namely, the use of screws to support the shear load as well as the tension load was unacceptable. The addition of shear pins relieved the screws from support of the shear load and thus solved the problem.

It is interesting to note that the basic structure was subjected to approximately 1-1/2 hr of vibration testing without sign of structural fatigue.

The required functional operations of the spacecraft were satisfactory.

Functional Tests

Satisfactory performance of the functional tests demonstrated that the functional and structural objectives of the spacecraft had, in general, been achieved.

The structural integrity of the spacecraft design was verified under functionally satisfactory tests of spinup, despin, experiment boom erection, solar paddle and mass boom deployment, and separation of the spacecraft from a simulated third stage.

A Lyman-alpha detector, which was in place during the spinup, showed no evidence of contamination from burning of the spinup motors.

Spacecraft separation from the third stage was satisfactory, with no indication of tipoff when one or both explosive release devices were fired.

After additional tests at GSFC, but without vacuum simulation, the escapement mechanism for the experiment booms was modified so that full deployment of booms was achieved at the minimum anticipated flight spin rate.

After twelve tests and with use of heat-treated springs, the stretch yo-yo system's performance satisfied flight requirements as verified during the final series of three tests. For the final series the springs had been treated for a high yield point. Improved performance resulted, but a slight yield occurred at 25 percent over nominal spin rate, constituting a slight deviation from design objectives. This system was used for flight.

Prototype and Flight Spacecraft

Vibration

The first vibration test of the prototype spacecraft revealed that the resonant frequency of the structure was 80 cps. The resulting acceleration levels up to 83g caused the photomultiplier tube of the cosmic ray experiment to fail.

It was eventually determined that the Dutchman was adversely affecting the structure's frequency, and stiffening of the Dutchman flanges raised the natural frequency to 95 cps.

At the same time the U.K. and the U.S. developed improved versions of the photomultiplier tube. Retest showed both these types of tubes to be satisfactory and demonstrated that the entire spacecraft could withstand design qualification levels of vibration.

The general behavior of the structure in vibration was relatively simple, as shown in Figures 11 through 16. In the thrust axis, one significant mode was established in the vicinity of 85 to 95 cps, beyond which the response declined to a 1-to-1 ratio at about 180 cps, with the responses above 180 cps being generally below the input. The main exception to the rule of isolation above 180 cps was in the case of the main equipment shelf (Figure 5), which experienced two sharply tuned resonances at 280 and 960 cps. Figure 17 depicts the Ariel I system with accelerometer locations.

The degree of attenuation at the 550 to 650 cps resonant-burning region was sufficiently great to limit the levels to reasonable values.

In the lateral direction two significant resonances were encountered: one at 35 cps, and the other at 120 cps. The former was so narrow in frequency as to be minor from a fatigue standpoint. The latter, which may or may not have been a manifestation of coupling with the fundamental thrust axis mode, was more severe but, because of the lower input level, probably represented less damage potential than the thrust axis test for all except equipment with particular susceptibility to lateral vibration.

Shunt Regulator

While the prototype spacecraft was being prepared for the first thermal-vacuum test, it became evident that the shunt regulator case needed to be electrically isolated from the basic structure. Redesign solved this problem.

During the first thermal-vacuum test the power dump transistors burned out because the overall shunt regulator circuit could not dissipate the energy required for spacecraft operation. The circuit was redesigned.

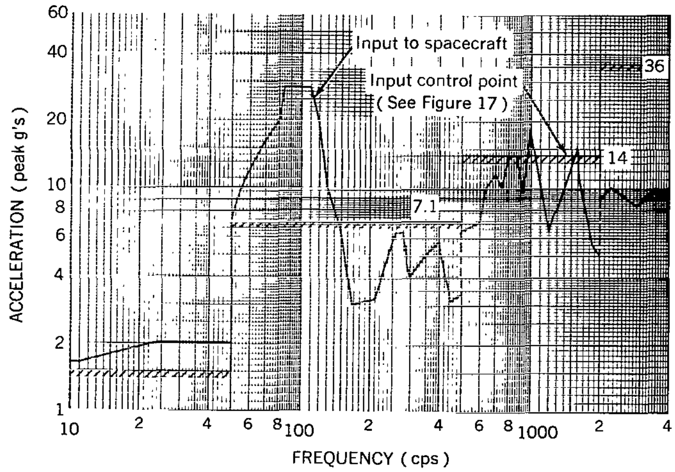

Figure 11—Flight unit 1 thrust axis, input to spacecraft.

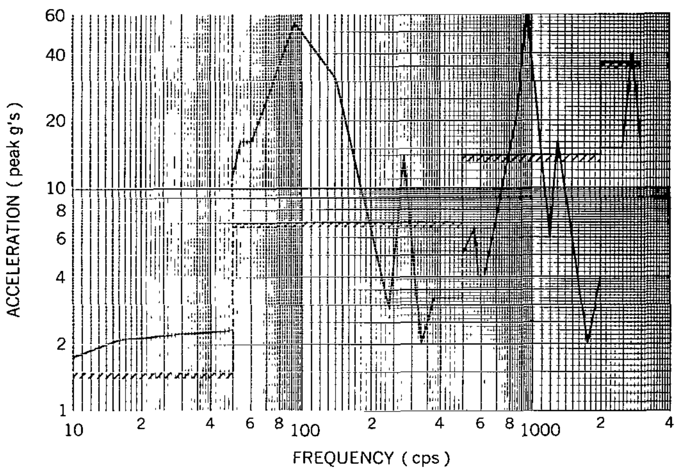

Figure 12—Flight unit 1 thrust axis, equipment shelf.

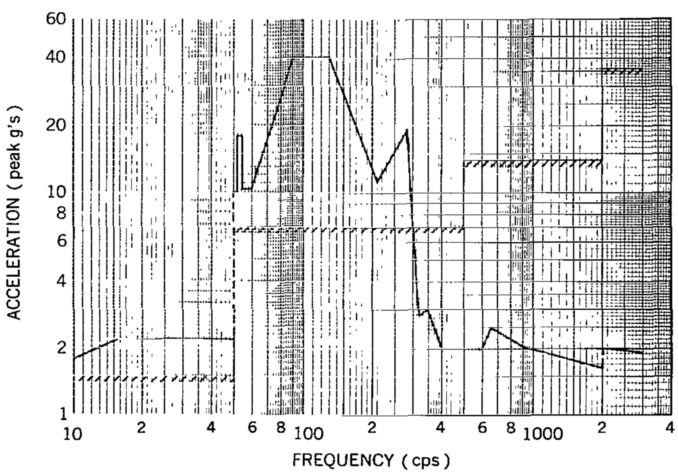

Figure 13—Thrust axis, input to cosmic ray experiment.

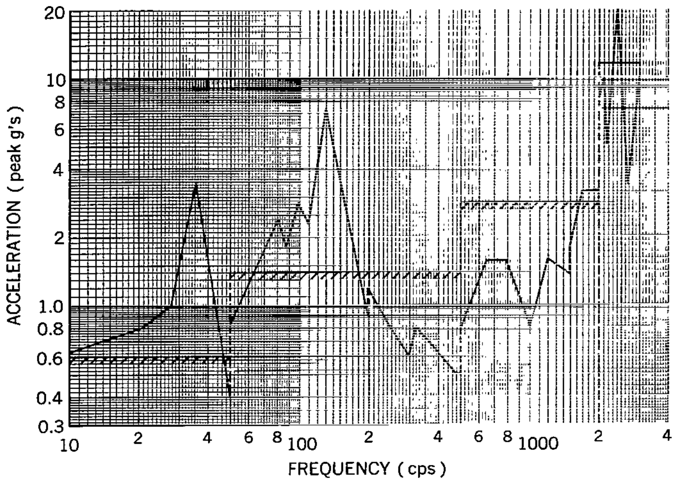

Figure 14—Flight unit 1 lateral axis, input to spacecraft.

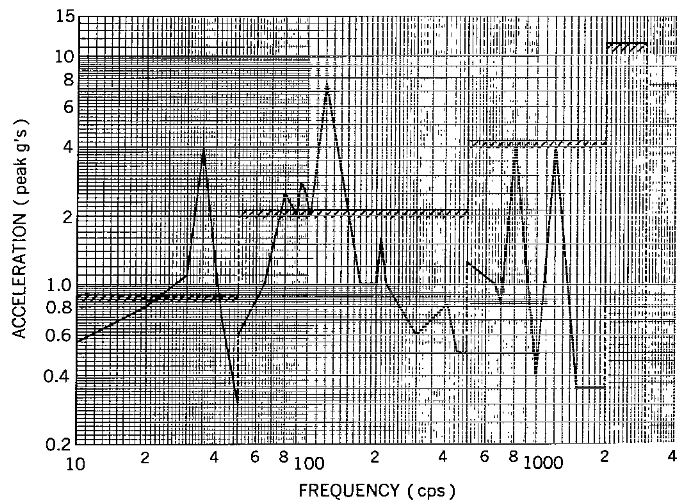

Figure 15—Flight unit 1 lateral axis, equipment shelf.

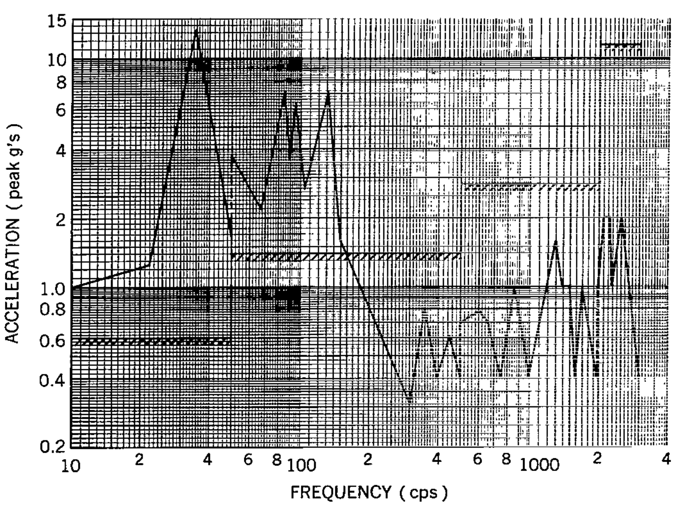

Figure 16—Flight unit 1 lateral axis, input to cosmic ray experiment.

During the second thermal-vacuum test, the batteries overcharged because the shunt regulator had no provision for current control. The redesigned circuit mentioned above also took care of this difficulty.

It was also observed that the power dump transistors were overheating during spacecraft operation. The trouble was traced to poor conductivity through the beryllium oxide washers into the solar paddle arms. Improved assembly technique solved this problem.

After solution of these design and assembly problems, the shunt regulators in the prototype and flight spacecraft performed satisfactorily in the required thermal-vacuum environments.

Tape Recorder

Problems and Solutions: Shorting of internal electrical leads to recorder case was solved by providing insulation between leads and case. A tape splice broke during testing; its weakness was caused by contact with tape lubricant at time of splice. A loose belt, causing failure in recording, was likewise caused by improper assembly techniques. Excess current drawn by the tape recorder resulted from seized bearings; the lubricant in the bearings was found to be unsuitable.

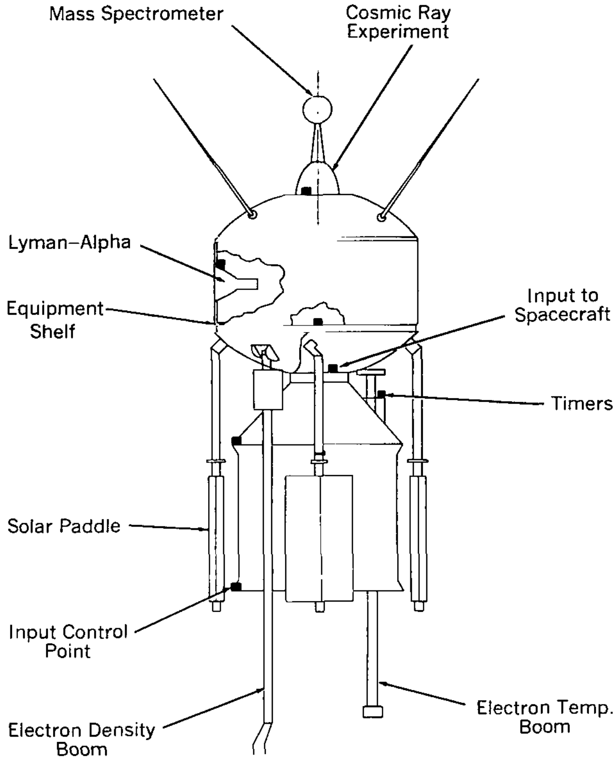

Figure 17—Typical Ariel I test configuration showing accelerometer locations.

Modification: As a result of the numerous tape recorder failures, causing excess current demands on the spacecraft, a current-sensing relay that would shut off the recorder when excessive current was drawn was placed in the powerline. Thus the overall operation of the satellite would be protected from excessive power drain in case of improper tape recorder operation.

Experiments

Electron Density: This experiment would not operate in a -10°C environment because it had not been designed for this condition. The problem was solved by application of thermal coatings to electronic cans on flight spacecraft. This modification changed the environmental requirement to 0°C, where operation was satisfactory.

Cosmic Ray experiment: The problem posed by the photomultiplier tube and its solution were discussed above under Vibration. During prototype thermal-vacuum, a Geiger counter tube failed; it was removed and was not replaced on flight spacecraft because it did not substantially affect experimental measurements.

X-Ray Experiment: The high voltage generator card failed during corona checks because of insufficient insulation. Additional insulation around the x-ray counters and high voltage leads at the counters and high voltage card was necessary. Another failure of the high voltage generator card was traced to leaks in the transformer on the card. Replacement of all transformers with a vacuum-impregnated type resulted in satisfactory operation during flight spacecraft testing.

Wiring Harness

The wiring harness had numerous operational failures because of incorrect assembly of connectors and numerous unshielded monitoring leads; this resulted in false indications of experiment malfunctions during prototype testing.

Redesign resulted in an improved harness with fewer monitoring leads, and improved assembly techniques also helped produce a superior harness for the flight spacecraft.

System and Test Evaluations

System

The results of the Ariel I environmental test program indicate that the spacecraft system and experiments were generally well-designed and reasonably reliable with the exception of the shunt regulator and the tape recorder.

After redesign and application of improved assembly techniques, the reliability of the shunt regulator was demonstrated by satisfactory operation during the flight acceptance tests and the retest of the prototype.

The operation of the tape recorder was erratic during both prototype and flight acceptance tests. Therefore, a circuit was designed to remove the tape recorder load from the spacecraft electrical system if a failure occurred. This fail-safe circuit resulted in an overall increase in the spacecraft's reliability.

Several other difficulties were attributable to improper assembly rather than to design shortcomings, and were discovered and corrected as a result of the test program.

Test

The test results indicate a minimum of difficulty traceable to human error in testing and therefore strongly imply that the test procedures were effective in avoiding damage to the spacecraft from improper handling and operation. In a broader sense, the effectiveness of the test program is supported by the successful operation in space of the Ariel I satellite.

Appendix A

Technical Description of Ariel I

General

The basic configuration of Ariel I (Figures A1 and A2), which must fit the shroud of the Delta rocket's spacecraft compartment, is that of a short fat cylinder 10-11/16 in. long and 23 in. in diameter.

Each end of the cylinder has a spherical section with an inboard terminator circle 23 in. in diameter and a smaller outboard terminator circle 8-7/16 in. in diameter. These spherical sections are 5-7/16 in. in diameter and 5-1/4 in. long, with an outer surface radius of 13-1/2 in. To this basic configuration are attached the various appendages necessary to support and conduct the spacecraft experiments.

The spin axis of the satellite is the central axis of the cylinder; this is also considered as the vertical axis. At the bottom of the spacecraft is a 9-3/8-in.-diameter third-stage separation flange, where an electron temperature gage and the tape recorder are installed.

Extending out horizontally from about midway up the lower spherical section, at 90-degree intervals around the circumference of the satellite, are four solar paddles.

Two 4-ft booms are offset 45 degrees circumferentially from the solar paddles. They are opposite to, and exactly counterbalance, each other and extend radially in the same horizontal plane. The end of one boom holds two circular condenser plates of the electron density sensor. Electronics

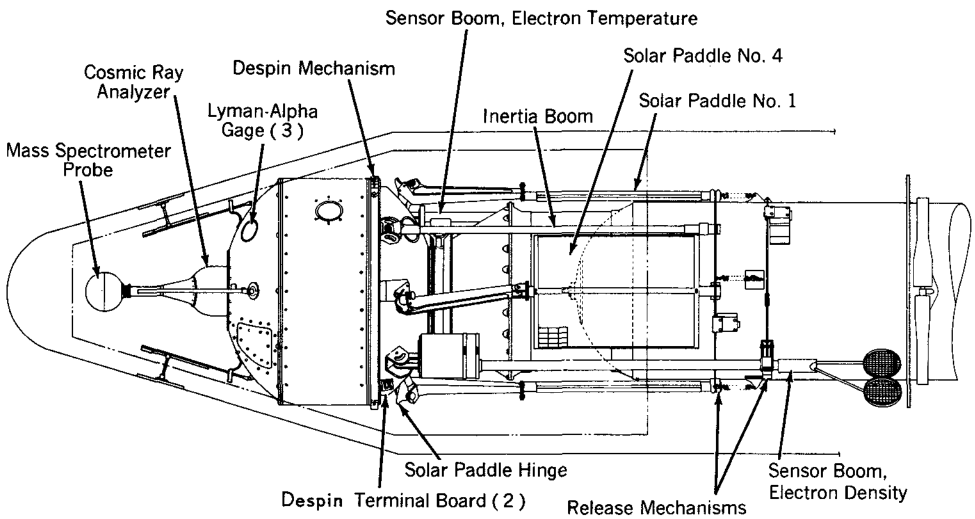

Figure A1—Ariel I in launch configuration.

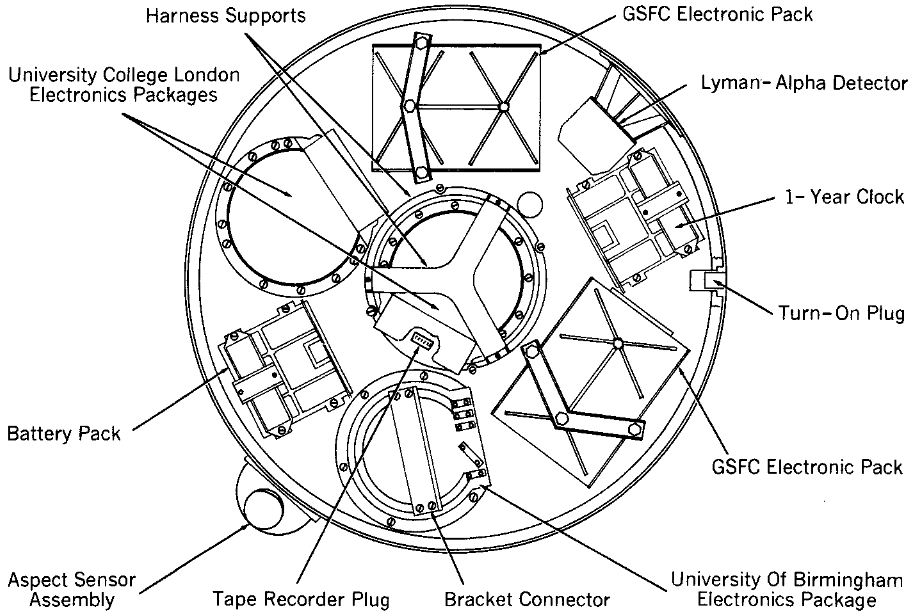

Figure A2—Top view of inside positioning for Ariel I electronic modules.

associated with this experiment are housed in a 4-1/4-in.-diameter by 6-1/4-in.-long cylinder mounted on the boom close to the spacecraft body. The end of the other boom holds a second electron temperature gage, whose electronics are located inside the spacecraft. A 3-1/2-in.-diameter hemispherical solar aspect sensor is located on the central cylinder section.

On top of the spacecraft in line with the spin axis is a 5-in.-diameter cylinder containing the cosmic ray Cerenkov detector. Above this, on a 4-in.-long conical section tapering from a 3-in. to a 1-in.-diameter is a 4-in.-diameter ion mass sphere.

Four antennas spaced circumferentially at 90 degrees and angling up at 45 degrees are mounted on the top spherical section. Three flush-mounted solar radiation (Lyman-alpha) gages are mounted on the satellite skin. Two proportional x-ray counters are located opposite the Lyman-alpha gages.

Structure and Mechanical Design

Basic structural materials in Ariel I are plastic-bonded fiber glass. The main or central body section is an epoxy-bonded monofilament-wound cylinder structure with a density of 0.071 lb/cu in. The upper and lower spherical sections (domes) are molded of the same materials and have the same density. The upper dome is 1/16 in. thick; the lower dome, 1/32 in. thick. The upper dome is bonded to an aluminum ring that, in turn, mates with an aluminum ring bonded to the mid-skin. The lower dome is assembled in sections.

Internally at the top of the upper dome is a 0.2-in.-thick aluminum disk, on which is centered an integrally machined aluminum cylinder 4 in. deep with a 7-in. inside diameter. Bonded to this central structure and the inside diameter of the upper dome are eight 1/16-in.-thick mat-molded epoxy-bonded fiber glass stiffening ribs. These ribs have holes in them for weight reduction.

At the top of the lower dome and at the bottom of the mid-skin is the instrument shelf, 0.080 in. thick and machined from 6061-T6 aluminum; on the underside of this shelf are eight integral stiffening ribs. A 7-in.-diameter base ring enclosing the tape recorder and providing a structure for bolting the spacecraft to the separation flange is mounted below the shelf. Bolted to the base ring and to the instrument shelf are six aluminum struts, also drilled out for weight reduction. Two struts support boom-mounted experiments, and the other four support the solar paddles.

The battery compartment is a simple mechanical container fitted with Teflon spacers. It is not pressurized but provides structural support to the sealed and pressurized nickel-cadmium batteries.

The flush-mounted x-ray sensors, light in weight, are bolted to the skin. Aluminum doublers mounted inside the skin at these points give additional stiffness. Lyman-alpha gages in the two domes are bolted to the skin and to brackets bolted to the central cylinder and the lower portion of the top shelf.

The Lyman-alpha sensor located on the cylindrical section is supported only by the skin, as is the aspect sensor. The top structure, including the Cerenkov sensor, is bolted to a plate on top of the central cylinder.

In the base of the spacecraft just below the tape recorder is an escapement device by which cabling is attached to the sensor booms to control their erection rate and timing. The "stretch yo-yo" despin system consists of two steel springs wound 1/2 turn each around the bottom of the cylinder just above the instrument shelf. At the end of each spring is a relatively heavy weight.

Approximate weight of the structure (without instrumentation, batteries, etc.) is 35 lb. Weight of the four antennas is 0.8 lb.

Thermal Design

A variation of 35°C is expected within the satellite structure over a period of 1 year because of variations in the amount of time spent in sunlight, and shadowing effects. Skin temperature may vary from about 20° to 60°C, depending on sun - spin-axis angle. The variation in temperatures of boom-mounted components due to sun - spin-axis angle changes may be somewhat greater than the temperature for the main structure because of nonspherical geometry and greater shading effects. Solar paddle temperatures should remain between +33°C and -63°C, the temperatures corresponding to the hottest and coldest combinations of spin axis, orbital plane, and orbital position locations with respect to the sun.

Ariel I uses the thermal coatings of evaporated gold with about 25 percent of the total surface area covered with a combination of black and white paints to achieve the desired absorptivity/emissivity ratio. This is a compromise between the experimenters' requirement for a conducting surface with a preference for a gold or rhodium over other metals, and a thermal requirement for maximizing the painted areas to minimize the tolerances of the absorptivity/emissivity ratio.

A special process was developed for applying the thermal coatings. Mirrorlike gold surfaces that should withstand heating to 250°F were obtained. First the surfaces were sanded, cleaned, and baked at 310°F for 1 hr. Then layers of varish, lacquer, paint, and metals were applied in the following sequence and baked at the temperatures and for the time intervals indicated: (1) sealing varnish, 300° F for 20 min; (2) lacquer, 290°F for 30 min; (3) conducting silver paint, 280°F for 13 hr; (4) electroplated copper, 1.5 mils thickness; (5) lacquer, 275°F for 30 min; (6) evaporated gold; and (7) four coats of black paint in longitudinal stripes, 250°F for 30 min.

Telemetry

General

The radiating system on Ariel I is a slightly modified crossed dipole or turnstile array mounted on the upper part of the spacecraft. The antennas are mounted on the upper part of the spacecraft, on the upper dome, directly over the solar paddles in a canted or V-shaped turnstile configuration. The telemetry transmitted operates at a frequency of 136.410 Mc and will be used for both data transmission and as a signal source for tracking. The output power to the antenna system is 250 mw with an overall transmitter efficiency of 35 percent. The transmitter is designed for use with a phase-lock-type receiving system in the ground stations.

The command receiver is a single-channel type operating on the standard NASA command frequency, which is amplitude-modulated by an assigned subcarrier tone. The standby power consumption is approximately 50 mw, and the sensitivity is -100 dbm. Upon interrogation of the command receiver from a ground station, information stored in the flight tape recorder is transmitted back to the ground station.

Telemetry Encoder

The telemetry system is a Pulsed Frequency Modulation (PFM) system. This is a particular form of time-division multiplexing in which the information being telemetered is contained in the frequency of a sequential series of 10-millisec pulses separated by 10-millisec intervals. The pulse frequency is derived from a set of pulsed subcarrier oscillators, each having a frequency range of from 5 to 15 kc.

Two encoders, termed the high-speed and low-speed encoders, are used. The encoders will accept transducer outputs from various experiments, commutate them, and produce a frequency proportional to the value of the parameter being measured in each experiment. The output from the high-speed encoder will modulate the transmitter directly (real-time data), while the output from the low-speed encoder will be recorded on a tape recorder for a complete orbit at 1/48 the information rate of the high-speed system. On command, the tape output will be played back at 48 times the recorded speed. Thus the output from both systems, when received at a ground station, will have the same bandwidth.

Some 66 separate parameters will be telemetered from the experiments. In addition, there will be three sync outputs generated in the encoder for a total of 69 parameters being telemetered.

The high-speed encoder output consists of 256 channels arranged in 16 frames with each frame in turn consisting of 16 channels. Since the blank-burst interval for each "channel" takes 20 millisec, a complete high-speed telemetry sequence of 256 channels will take 5.12 sec.

The low-speed encoder output into the tape recorder consists of 32 channels arranged in two frames with each frame in turn consisting of 16 channels. Since the burst-blank interval for each channel is 0.96 sec (e.g., 1/48 that of the high-speed system), a complete low-speed telemetry sequence of 32 channels will take 30.72 sec.

Recorder

General

The spacecraft recorder is designed to store encoded data for time periods up to 100 min. Input to the recorder is from a low-speed encoder that furnishes data signals at 1/48th of the rate at which data are transmitted in real time. Recorder playback, initiated on receipt of ground-station command, will be at 48 times record speed—thus, the transmitted information rate will be identical for recorded data and real-time data.

The recorder contains 150 ft of special lubricated tape in an endless-loop configuration. Upon command, a 2-sec burst of 321 cps will be made to provide a distinctive time-reckoning mark (this burst will also be transmitted to signify successful command initiation). After the 2-sec interval, the recording ceases and the recorder will play back for 2.1 min, and then commence recording again.

Weighing 2-1/2 lb, the recorder is 7 in. in diameter and 3 in. high. Total power input required is 0.7 watt.

Tape Recorder Programmer

This programmer provides command means by which the tape recorder may be placed in the playback mode to allow extraction of orbital data. A precision reference frequency source is also an integral part of the programmer.

One-Year Timer

Two long-term electrolytic timers are provided in a redundant system to permit radio silencing of the satellite, nominally 1 year after launch.

The time interval is set by the selection of current allowed to flow in each timer. The switch on each timer is normally closed and connected in parallel so that the last timer to actuate determines the timing period and turns off the transmitter.

Power

Power System

The main elements of the power system are:

1. Four solar paddles
2. Shunt regulator and battery charge current limiter
3. Battery switching network
4. Batteries
5. Undervoltage system
   (a) Undervoltage sensing circuit
   (b) Timer for shutdown of satellite for 18 hr for battery recharging

The solar paddle outputs are combined and connected to a voltage regulator that is in parallel with the solar cells and the remaining power system.

The solar cells charge the batteries and supply all operating power to the spacecraft during the time the spacecraft is in the sunlight.

The regulator limits the system voltage and the battery charge current from the solar cells to values below which battery internal pressures rise as a result of generation of hydrogen.

The battery switching network is in series with both batteries, and the combination is in parallel with the solar cell paddles and shunt regulator.

The battery switching network connects either Battery A or Battery B to the load.

The hold-off relay is connected in the main powerline in series with the spacecraft subsystems. It will provide remote control of satellite turn-on.

The undervoltage system is connected in the main powerline in series with the spacecraft subsystems. It will turn off the satellite if the load voltage of Battery A and the load voltage of Battery B go below a predetermined value. During the shutdown period, all power output of the solar cells is connected to the battery switching network that charges the battery on line and trickle-charges the other battery.

Power Supply

The power supply for Ariel I consists of photoelectric converters; these generate electrical energy used to charge electrochemical storage batteries.

The solar converters are four fixed-position paddles, measuring approximately 1 ft by 1 1/4 ft, having solar cells covering both sides. The planned output of the total array is slightly in excess of

10 watts at 15 volts. In order to prevent overcharging of the storage batteries, a zener shunt regulator and battery charging current limiter was designed by GSFC to limit the voltage to 14±2 volts.

The nickel-cadmium storage battery is a 6-amp hr size. Two 10-cell batteries are employed, one on standby service; the nominal voltage is 12 volts. GSFC designed an electronic circuit that should automatically keep the load on the battery in the highest state of charge and best operating condition.

Because of the different voltages required by the various experiments within the spacecraft, four dc to dc transistorized electrical converters are used. Three of these converters were designed and manufactured at GSFC.

Separation System

For reliability, the separation system is composed of two completely independent and isolated systems with either one having the capability for performing all required functions; see Figure A3.

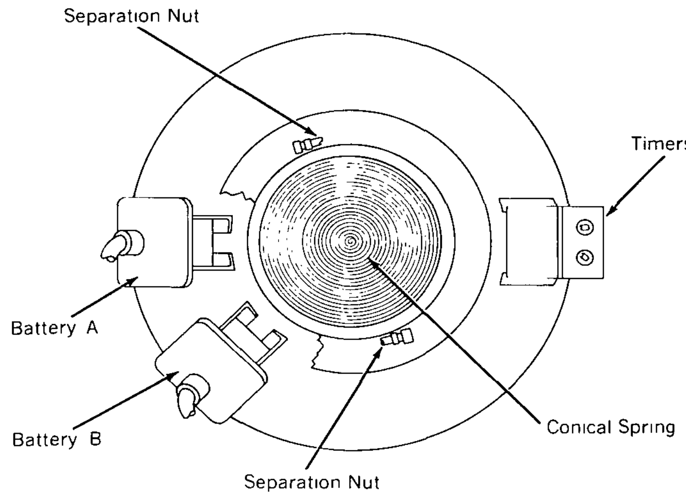

Figure A3—Top view of separation system.

The major components of both systems are mounted on a hollow frustum. Circuitry extends through the flyaway connector at the spacecraft separation system interface into the spacecraft for release of the despin mechanism. In addition, circuitry extends down along the Dutchman and the third-stage rocket motor to the experimental boom and initial boom release mechanisms.

Each independent system consists of the following elements: (1) battery, eight silver-zinc cells, 1/2 amp hr; (2) timer, electronic; (3) guillotine; (4) pin puller; and (5) separation nuts, powder cartridge.

Appendix B

Test Chronology

Structural Model

Balance Operations

Time: April 27, 1961.

Weights: Initial unbalance - 1.5×10^7 dyne-cm (218 oz-in.)
Addition - 0.68 kg (1.5 lb)
Residual unbalance - 4.4×10^6 dyne-cm (61.8 oz-in.)

Vibration

Time: May 2, 1961.

Discrepancy: Screws sheared in several locations.

Acceleration

No problems.

Final Vibration

No problems.

Spin

No problems.

Prototype Spacecraft

Balance Operations

Time: September 15-19, 1961.

Weights: Initial spacecraft weight - 105.6 lb*
Balance weight added - 1.0 lb
Final spacecraft weight - 106.6 lb

Spacecraft Initial Unbalance: Static - 176 oz-in.
Dynamic - 448 oz-in.^2

*Not including the weight of the booms and paddles (weight as balanced).

Residual Unbalance: Static - 5 oz-in.
Dynamic - 25 oz-in.^{2}

The battery switching network was not installed until the completion of the balance operations. The spacecraft unbalance on completion of the balance operations was calculated to be:

Static - 82 oz-in.
Dynamic - 273 oz-in.^{2}

Results: The planes selected for balance weight additions were found to be suitable.

Spin Test

Time: September 19, 1961.

Test Conditions: 225 rpm for 1 min
150 rpm for 30 min

The specification was not followed exactly because the balancing machine could not be operated below 150 rpm.

Difficulty:

Problem - Spacecraft would go into undervoltage upon insertion of turn-on plug.

Cause - Low battery voltage.

Temperature and Humidity Tests

Instrumentation and Test Setup: September 20-22, 1961.

Transmitter and receiver leads were interchanged. Investigation showed that the two leads were mislabeled in the spacecraft harness (Sept. 22, 1961).

Satisfactory checkout at +25°C (Sept. 22, 1961).

Cold-Storage Temperature Test: September 22, 1961.

Spacecraft subjected to -30°C storage test (nonoperating) for 6 hr.

Ambient Temperature Checkout: September 23, 1961.

Satisfactory spacecraft checkout at +25°C after -30°C storage test.

Ground Station Problem: September 23, 1961.

Sanborn recorder was functioning unsatisfactorily. Also, the oscilloscope used as additional aid in evaluation was not operating properly.

Hot-Storage Temperature Test: September 23, 1961.

Spacecraft subjected to +60°C storage test (nonoperating) for 6 hr.

Ambient Temperature Checkout Following +60°C test:

Connector problem on the prime converter. Smaller connector standoffs were required for proper connection. Also, additional thermocouples were installed within the electronic modules. (Sept. 24, 1961).

Faulty connector for transmitter lead in spacecraft harness was replaced (Sept. 25, 1961).

Satisfactory checkout at +25°C completed September 25, 1961.

Cold-Operation Temperature Tests:

The spacecraft was stabilized at +5°C nonoperating. The spacecraft was then operated and a satisfactory checkout completed (Sept. 25, 1961).

The spacecraft, while nonoperative, was then stabilized at -10°C. The spacecraft was operated following stabilization, and the following problems were observed:

> Electron density experiment was gating at random (improper). The output indicated maximum (6-7 volts) and could not be varied when the capacitance of the exciter was varied.
>
> Optical aspect readout not correct. Later investigation showed this to be caused by insufficient closing time of microswitches in the exciter electronics (readjusted at a later date).
>
> X-ray experiment showed erratic counting of the reference number. No final solution as of October 1, 1961.
>
> Cosmic ray experiment count was improper.
>
> Tape recorder was operating but at a slower rate. Also, it was drawing excessive current.

Ambient checkout at conclusion of the cold-operation temperature tests, September 25, 1961.

Satisfactory checkout at ambient temperature. All experiments recovered. Therefore, no repairs were made.

Hot-Operation Temperature Test:

The spacecraft, while nonoperative, was stabilized at +50°C. The spacecraft was operated following stabilization, and the following problem was observed:

> Cosmic ray count was not proper. A defective Geiger tube was suspected.

Ambient Checkout Following 50°C Test: September 28, 1961.

Cosmic ray problem as noted in hot-operation test.

The decision was made to continue with the repeat of the cold-operation temperature test with this problem present.

Cold-Operation Temperature Test Rerun: September 28, 1961.

Spacecraft temperature stabilized at +5°C prior to checkout. The following problems were noted:

> The tape recorder was drawing excessive current. Electron density operation was not normal. Cosmic ray degradation continued. Temperature chamber temperature was set at 0°C, and the temperature of the monitored points of the satellite was allowed to stabilize with the satellite operating.
>
> Bulb in optical aspect exciter burned out. Tape recorder and electron density problems continued. Cosmic ray degradation continued.

The chamber temperature was lowered to -5°C with the spacecraft operating during the temperature change. All problems in cold-operation temperature test rerun continued. No new problems developed.

The chamber temperature was lowered to -10°C with the spacecraft operating. No new problems developed.

Ambient Checkout Following -10°C test: September 28, 1961.

Satisfactory checkout prior to humidity. All experiments recovered from -10°C except cosmic ray (defective Geiger tube).

Humidity Test:

Spacecraft exposed to 95 percent RH at 30° C for 24 hrs and then checked out (Sept. 29, 1961).

X ray showed excessive current at 95 percent RH. Recovered when chamber reached 80 percent RH (5 min).

Several test points were out of tolerance until 42 percent RH was reached by chamber (10 min). Then, these parameters reached the values of minimum acceptable tolerance.

Ambient Temperature Checkout Following Humidity Test: September 29, 1961.

Satisfactory checkout except for cosmic ray, which had a defective Geiger tube.

Vibration Tests

Time: October 10-13, 1961.

Thrust Axis Test (Tall Fixture): October 10, 1961.

Sinusoidal Sweep Frequency Test (10-250 cps) —

1. Electron density disk broke off during the 83 cps resonance:

   Cause: Previously damaged during handling.

   Solution: An epoxy cement was used to attach disk for remainder of test (immediate). New disks will be fabricated and attached prior to re-test (final).

2. Tape recorder would not operate during checkout at end of test:

   Cause: Blown fuse (in spacecraft).

   Solution: Fuse replaced by jumper wire. (Tape recorder was not operating during tall fixture-thrust axis test.)

3. Cosmic ray count rate shifted at 83 cps structural resonance:

   Cause: Failure in photomultiplier tube.

   Solution: (a) Reduce amplification in system that occurs during resonance.
   (b) Redesign tube to withstand higher acceleration levels.
   (c) Final solution has not been proven as of January 16, 1962.

Random Test (psd = 0.07 g^2/cps; frequency band, 20-250 cps) —

No problems noted.

<u>Thrust Axis Test (Short Fixture)</u>: October 11, 1961.

Sinusoidal Sweep Frequency Test (250-3000 cps) —

No problems encountered.

Random Test (psd = 0.07 g^2/cps; frequency band, 250-2000 cps) —

No problems encountered.

<u>Transverse Test, X'-X' Axis (Tall Fixture)</u>: October 12, 1961.

Sinusoidal Sweep Frequency (10-150 cps) —

Intermittent operation of the electron density experiment.

Random Test (psd = 0.07 g^2/cps; frequency band, 20-150 cps) —

Electron density problem continued.

Resonance Test (sinusoidal sweep, 550-650 cps) —

Electron density problem continued.

<u>Transverse Test, Y'-Y' Axis (Tall Fixture)</u>: October 12, 1961.

Sinusoidal Sweep Frequency (10-150 cps) —

Electron density problem continued.

Random Test (psd = 0.07 g^2/cps; frequency band, 20-150 cps) —

Electron density problem continued.

Vibration experiment simulated battery box separated from its mounting brackets.

Cause: Failure of six spot welds used to attach battery box to brackets.

Solution: (a) A continuous weld was used for attaching box to brackets.
(b) Additional bolts were employed to attach the battery box to the Dutchman.

Antenna failure.

Cause: Antenna was previously damaged.

Solution: Antenna replaced.

Transverse Test, Y'-Y' Axis (Short Fixture): October 13, 1961.

Sinusoidal Sweep Frequency (150-2000 cps) —

No problems encountered.

Random Test (psd = 0.07 g^2/cps; frequency band, 150-200 cps) —

No problems encountered.

Resonance Test (sinusoidal sweep, 550-650 cps) —

No problems encountered.

Transverse Test, X'-X' Axis (Short Fixture): October 13, 1961.

Sinusoidal Sweep Frequency (150-2000 cps) —

Electron density problem continued.

Resonance Test (sinusoidal sweep, 550-650 cps) —

Electron density problem continued.

Random Test (psd = 0.07 g^2/cps; frequency band, 150-2000 cps) —

Tape recorder played back without initiation command during 600 cps resonance test.

Cause: (a) Test repeated twice to determine whether any interactions had inducted playback—none were found.

(b) Command frequency and tone for playback of Ariel I (S-51) and the Atmospheric Structures Satellite (S-6) were found to be the same (playback could not be traced to S-6 operation).

Solution: Ariel I command playback tone changed.

Post Vibration Test Checkout: October 14, 1961.

Mass spectrometer inner sphere unscrewed, causing several electrical contacts to shear.

Cause: Lock-tite had not been applied.

Solution: Closer pre-test inspection required.

Blown fuses in x-ray and aspect -6.5 volt lines and mass spectrometer 24 volt line.

Cause: Suspect switching transients when system is turned off and on.

Solution: Remove all fuses from spacecraft.

Acceleration

On October 17, 1961, thrust axis acceleration test of 25g was conducted at NRL's Chesapeake Bay Annex.

Problem at Test Site:

Tape recorder would not play back on RF command.

Cause: Defective switch on command control panel (general support equipment).

Solution: Replaced switch.

Checkout after Test:

Spacecraft performance data were magnetically recorded before, during, and after the test and were returned to GSFC for evaluation on October 18, 1961. Since the spacecraft had not been repaired after the vibration test, the same problems noted were observed to have been present before and after the test. The following additional problems were noted:

Wiring harness—excess loading on -2.7 volt line.

Cause: Two loads on one supply line.

Solution: Separation lines installed in harness to supply each load.

Undervoltage detector circuit—initiation of both oscillators did not always occur (redundant system).

Cause: Design problem.

Solution: The wiring circuit was changed.

Shunt regular box was installed but not connected to circuit, since power transistors were not available.

Command receiver—length of modulation burst became excessive (5 sec)—normal, 2 sec.

Cause: Defective diode.

Solution: Replaced.

Cosmic ray operation not proper.

Cause: Defective Geiger tube (large tube); first noted at +55° C temperature test.

Solution: Removed tube from circuit.

Leaky tape recorder.

Cause: Surface damage produced poor seal.

Cause: Sensor shorted to ground.

Solution: Repaired.

Electron temperature probe no. 1 would not sweep (Nov. 2, 1961).

Cause: Faulty micro-dot cable at connector from electronics to sensor.

Solution: Replaced cable.

Encoder clock would not give proper period pulse on low speed (Nov. 2, 1961).

Cause: Sensitive to position.

Solution: Replaced with spare clock. Also, low-speed and high-speed encoders updated at this time.

Ground Station (Nov. 2, 1961).

Cause: Decoder not proper.

Solution: Repaired.

Spare encoder clock not operating properly (Nov. 3, 1961).

Cause: Capacitor missing from the card.

Solution: Replaced encoder clock with a pre-prototype clock.

Chamber Evacuation: November 5, 1961.

Spacecraft was operated and monitored until chamber pressure of 1×10^{-4} mm Hg was achieved.

X ray showed rapid count at 70 mm Hg (Nov. 5, 1961).

Cause: Arcing of -1600 volts to sensors.

Solution: Test continued. Experiment operated properly at 2×10^{-4} mm Hg. (Special tests performed at later date.) EHT Card no. 503.

Downtime: November 5-9, 1961; for trouble-shooting of x ray.

Setup: November 9, 1961.

Faulty thermistor in UCL-2.

Cause: Open lead in monitor board.

Solution: Corrected November 11, 1961.

Chamber Evacuation: November 9, 1961.

Operation of the spacecraft appeared normal except as noted in "Setup" section. Also, the x-ray experiment was not operated during the chamber evacuation to 1×10^{-4} mm Hg.

Stabilization of the Spacecraft (not operating) at -10°C: November 9, 1961.

Low-speed oscillator malfunctioned.

Cause: Shorted gate transistor. This condition occurred during the initial chamber evacuation but was not noted until the low-speed data were reduced at a later date.

Solution: Corrected on November 11, 1961.

Downtime: November 9-11, 1961; for correction of problem noted above.

Setup: November 12, 1961.

Low-speed encoder malfunctioned (Nov. 12, 1961).

Cause: Open lead in printed circuit board of the low-speed matrix.

Solution: Repaired.

Cosmic ray would not gate properly (Nov. 13, 1961).

Cause: Three broken coaxial (miniature) leads.

Solution: Repaired.

Chamber Evacuation: November 13, 1961.

Spacecraft operated during pumpdown until chamber pressure of 1×10^{-4} mm Hg was achieved. Operation appeared normal; however, x-ray experiment was not operated during evacuation.

-10°C Vacuum Soak: November 13-21, 1961.

X-ray experiment would not turn on (Nov. 15, 1961).

Cause: A temperature-sensitive transistor within the EHT generator card was suspected.

Solution: Experiment operated after 1 hr, 20 min warmup period.

Cosmic ray not operating properly (Nov. 15, 1961).

Cause: Temperature-sensitive binary counter.

Solution: No action; cleared at -2.0°C.

X-ray experiment malfunctioned (Nov. 16, 1961).

Cause: Failure of -1600 volts to sensors.

Solution: Test continued. Additional potting applied at connector of -1600 volt line and sensor potted. Also, a vacuum-impregnated high voltage transformer was installed.

Shunt regulator would not operate (Nov. 17, 1961).

Cause: Dumping transistors burned out.

Solution: Redesign of circuit required.

+55°C Vacuum Soak: November 21-25, 1961.

Problems as indicated in preceding three paragraphs continued.

Tape recorder malfunctioned (Nov. 23, 1961).

Cause: (a) Broken splice in tape.
(b) Burned ground wire.
(c) Motor bearing not pre-cooled.
(d) Pressure switch wired backward.

Solution: Repaired.

Cosmic ray not operating properly (Nov. 23, 1961).

Cause: Faulty Geiger tube (big).

Solution: Cut out of circuit, as no replacement was available.

Lyman-alpha not operating properly (Nov. 23, 1961).

Cause: Diode open.

Solution: Repaired.

Optical aspect drawing excessive current (Nov. 23, 1961).

Cause: Shorted transistor.

Solution: Repaired.

U.K. converter—excessive temperature (Nov. 24, 1961).

Cause: Inadequate heat sink.

Solution: Improve heat sink.

Downtime: November 25 to December 6, 1961; for correction of problems November 12 on.

Setup: December 6, 1961.

The following modifications were accomplished:

1. Undervoltage changed to 18 hr at 11.1 volts from 24 hr at 10.5 volts.
2. Shunt regulator redesigned.
3. One-year timers installed.

X-ray drawing excessive current, no high voltage at the sensors (Dec. 7, 1961).

Cause: Faulty card.

Solution: Replaced.

Replaced pre-prototype encoder clock by first encoder clock, which had been repaired (Dec. 8, 1961).

Chamber Evacuation: December 9, 1961.

X-ray experiment malfunctioned (Dec. 9, 1961).

Cause: Arcing of -1600 volts at 7 mm Hg.

Solution: Removed EHT card from system and installed dummy card: New card available approximately February 1, 1962; x-ray experiment will be qualified at this time.

Cosmic ray not operating properly (Dec. 9, 1961).

Cause: Ground station.

Solution: Repaired.

+55°C Vacuum Soak: December 10-13, 1961.

Cosmic ray not operating properly (Dec. 10, 1961).

Cause: Ground station.

Solution: Repaired.

Optical aspect drawing excessive current (Dec. 10, 1961).

Cause: Faulty capacitor in card.

Solution: Repaired.

U.K. converter temperature excessive—65° C (Dec. 10, 1961).

Cause: Heat sink not properly installed.

Solution: Installation corrected on December 21, 1961.

Tape recorder pressure indicated low (Dec. 10, 1961).

Cause: Faulty pressure switch.

Solution: Use of pressure switch abandoned.

Undervoltage detector turned spacecraft off at 11.8 volts when undervoltage was set at 11.1 volts (Dec. 11, 1961).

Cause: Suspect that undervoltage detector was not properly temperature-compensated.

Solution: Power system and undervoltage detector presently being investigated for possible design change.

Lyman-alpha operation not proper (Dec. 11, 1961).

Cause: Potting found in right-angle connector at the Lyman-alpha sensor.

Solution: Replaced connector.

Battery life questionable (Dec. 11, 1961).

Cause: Charging rate excessive.

Solution: Possible redesign of power system may be required.

Shunt regulator allowing charging of batteries at excessive rate (Dec. 13, 1961).

Cause: Suspected design problem in shunt regulator.

Solution: Replaced shunt regulator by a redesigned unit.

Electron density (SEL) data store showed no erasure (Dec. 13, 1961).

Cause: Unknown.

Solution: Special temperature test showed operation normal.

Downtime: December 14-22, 1961; for correction of items occurring in +55°C vacuum soak.

Setup: December 22, 1961.

Electron temperature not operating properly (Dec. 22, 1961).

Cause: Faulty connector at dummy load.

Solution: Replaced.

-10°C Vacuum Soak: December 26-29, 1961.

Electron density not operating properly. Large shift in null.

Cause: Sensitive to low temperature.

Solution: (a) Heat lamps on boom electronics were used to increase temperature. Operation became normal at approximately 0°C.

(b) Study is being conducted to determine whether this item should have a thermal coating applied.

Tape recorder would not turn off during undervoltage (Dec. 27, 1961).

Cause: Test panel wiring.

Solution: No action.

Downtime: December 29 to January 2, 1962; test suspended because of holiday.

+47°C Vacuum Soak: January 2-5, 1962.

Battery temperature excessive (Jan. 4, 1962).

Cause: Shunt regulator is designed to limit the voltage to the batteries at 13.5 volts. When the batteries are being charged, their temperature increases to cause a decrease in impedence and voltage and an increase in the current.

Solution: Shunt regulator was redesigned to incorporate current-limiting features as well as voltage regulation. In addition, the bottom of the battery pack was machined flat to form a better thermal contact with the satellite instrumentation shelf.

Tape recorder drawing excessive current (Jan. 4, 1962).

Cause: Lubricant crystallized.

Solution: During the remainder of the +47° C test and the subsequent 30 and 135 degree solar aspect tests, the tape recorder was turned off.

30° Solar Aspect:

Shunt regulator produced a 1-Mc oscillation across the batteries when dumping circuit was operating (Jan. 6, 1962).

Electron density was not operating properly. Single period of 138 μsec was present instead of normal two sawtooth pulses (Jan. 6, 1962).

Cause: Suspect temperature gradient from sensor to electronics on boom.

Solution: Operation became normal as temperature of the electronics was increased.

Electron temperature not operating properly (Jan. 7, 1962).

Cause: Open condenser in sensor located in boom.

Solution: Repaired.

135° Solar Aspect: January 7-8, 1962.

No new problems developed. Previous problems continued.

Downtime: January 10 to February 10, 1962; all satellite deficiencies were corrected during this time.

Vibration Retest

Time: February 15, 1962.

Results: Satisfactory.

Thermal-Vacuum Retest

Time: March 7-12, 1962.

Results: Satisfactory.

Prototype Vibration Experiment and Separation System

Vibration

Time: March 10, 1962.

Results: Mechanical timers in separation system not accurate enough.

Temperature

Time: March 10, 1962.

Results: Satisfactory.

Humidity

Time: March 11-12, 1962.

Results: Satisfactory.

Thermal-Vacuum

Time: March 17-18, 1962.

Results: Satisfactory.

Flight Unit 1

Balance

Time: January 11-15, 1962.

Simulated Items: Cosmic ray experiment.
Shunt regulator circuit.
Tape recorder current limiter.

Vibration

Time: February 7-11, 1962.

Results: Flight versions of above simulated items installed prior to vibration. No problems.

Thermal-Vacuum

Setup and Checkout: February 16, 1961.

Cold Test: February 17-21, 1962.

Cosmic ray experiment would not turn on at -8°C. Operation became normal at +10°C.

X-ray experiment was not gating properly.

Cause: Faulty diode.

Solution: Replaced (action taken subsequent to thermal-vacuum test).

Electron energy no. 2 malfunctioned.

Cause: Open capacitor.

Solution: Replaced (action taken subsequent to thermal-vacuum test).

Shunt regulator power transistors operated at excessive temperatures.

Cause: Inadequate heat sink.

Solution: Modified method of mounting.

Hot Test: February 23-25, 1962.

Last three items in cold test (above) continued during this test. Electron density questionable.

Cold Test: February 26-27, 1962.

Items in paragraph above continued.

Corona Check: February 27, 1962.

No new problems.

Flight Unit 2

Balance

Time: February 5-8, 1962.

Simulated Items: X-ray experiment stack (UCL-2).
Cosmic ray experiment.
Tape recorder.

Vibration

Time: February 21-24, 1962.

Setup: Flight versions of above simulated items installed prior to vibration.

Results:

Cosmic ray experiment failed during random vibration—tall-fixture thrust direction.

Cause: Failure of weld in photomultiplier tube.

Solution: Spare cosmic ray unit was vibration-qualified and installed in the system prior to thermal-vacuum.

Mass spectrometer operation was intermittent.

Antenna problem.

Cause: Loose micro-dot connector.

Solution: Tightened.

Thermal-Vacuum

Setup and Checkout: March 1-3, 1962.

Mass spectrometer malfunctioned.

Cause: Broken pin in sphere; probably occurred during vibration testing.

Solution: Replaced sphere.

Tape recorder dc control unit questionable.

Cause: Circuit design problem.

Solution: Unmodified unit installed. All units are being modified to correct design problem and reduce sensitivity.

Corona Test: March 3, 1962.

No problems encountered.

Cold Test: March 5-8, 1962.

Tape recorder malfunctioned.

Cause: Faulty idler bearing.

Solution: Replaced bearing.

Electron density questionable.

Cause: Under investigation.

Solution: Experiment was being operated at an unrealistically low temperature during test.

Hot Test: March 10-13, 1962.

No new problems. Tape recorder operated satisfactorily for 10 hr during this test, at which time malfunction reoccurred.

Flight Unit 1 Vibration Experiment and Separation System

Temperature and Humidity

Time: March 11-12, 1962.

Results: Satisfactory.

Vibration

Time: March 17, 1962.

Results: Mechanical timers in separation system not accurate enough.

Thermal-Vacuum

Time: March 17-18, 1962.

Results: Satisfactory.

Flight Unit 2 Separation System

Temperature and Humidity

Time: March 11-12, 1962.

Results: Satisfactory.

Thermal-Vacuum

Time: March 17-18, 1962.

Results: Satisfactory.

Vibration

Time: March 21-24, 1962.

Results: Mechanical timers in separation system inaccurate.

Appendix C

Description of Flight Vibration Experiment

The vibration telemeter installed in the Ariel I Dutchman is intended to provide measurements of flight vibration levels imposed on the spacecraft during burning phases of the Thor-Delta vehicle. Three mutually orthogonal accelerometers located on the Dutchman motor attachment ring will provide these data for the thrust, pitch, and yaw axes. An additional channel on this telemeter has been allocated for a combined measurement of solar aspect, optical contamination, and third-stage pressure switch closure.

The vibration telemeter is a four-channel FM/FM system employing a solid-state transmitter with 1.8 watts output on 240.2 Mc. An RF bandpass filter suppresses spurious radiation to IRIG specifications. Antennas are a pair of quadraloops, diametrically positioned and bonded to the third-stage casing near the after end.

The vibration pickups are small piezoelectric accelerometers operating into separate charge amplifiers. System frequency response is limited by a low-pass filter with a cutoff frequency of 600 cps and a rolloff of about 9 db/octave. Thrust, pitch, and yaw vibration data modulate voltage-controlled oscillators on IRIG bands E, C, and A respectively. Solar aspect, contamination, and third-stage pressure switch closure data signals are mixed and modulate a voltage-controlled oscillator on IRIG band 13. A three-point ground-controlled voltage calibrator in the package provides 0, 2.5, and 5 volt levels to all voltage-controlled oscillators in parallel prior to launch. Power for the vibration telemeter package is provided by a pressurized pack of 20 Ag cells. Current drain is approximately 800 ma. Power for the aspect and contamination sensors is supplied by two 2-volt Hg cells.

Appendix D

Chronology of Atlantic Missile Range Operations, 1962

March 27:

It was found that the separation system and solar paddle arms (in folded position) interfered with each other. Removal of spacers on battery box and slight re-routing of separation system wiring harness near electronic timers resolved the problem.

This incident shows the need for mechanical interface checks between all major assembled units prior to shipment to Cape Canaveral.

March 29:

Fitting of Flight Unit 2 to third stage was checked and found to be satisfactory.

Final assembly of Flight Unit 1 on third stage was completed.

March 30:

During checkout a bad contact in one of the dipole motor flyaway pins (between the spacecraft and separation system) was found. Spacecraft was removed to clean contacts.

The escapement mechanism for the electronic booms was adjusted.

The solar paddles were installed but were found to be incorrectly located, and were re-positioned.

March 31:

Third stage and spacecraft were placed on balance machine to start balancing. In the afternoon the electron density boom grids were inadvertently damaged.

April 2:

The electron density boom from Flight Unit 2 was substituted for the damaged one.

The spacecraft was removed from the balance machine and taken to the antenna range for boom calibration. During reassembly for balance the escapement mechanism for boom erection was found to be defective and was replaced.

April 3:

The spacecraft weighed 135.50 lb, and the separation ring weighed 0.46 lb; this weight plus the Dutchman and separation system gives a total of 177.9 lb.

April 4:

Balancing was completed. Spacecraft, third-stage combination was placed in handling container in preparation for installation on pad. Installation was accomplished at 1500 EST.

April 5 and 6:

Integrated system checkout verified satisfactory performance of spacecraft.

April 7:

Spacecraft was on gantry with launch vehicle. Strip coat was removed in afternoon after rain stopped.

April 8:

Thermal coatings were touched up, and batteries were charged.

April 9:

Complete spacecraft checkout was satisfactory.

April 10:

Countdown proceeded until T-6 minutes when 2nd stage failed to pressurize. Launch was scrubbed after hold of 4-1/2 hr.

April 11:

Nitrogen pressurization tank of second stage was found deficient. Launch was rescheduled for April 25, 1962, with a new second stage. The spacecraft was to be removed from third stage for recalibration. The vibration experiment also was to be recalibrated.

Reassembly and balance was scheduled for April 18-20, and mating with third stage was to be at 0800, April 21.

April 12:

Spacecraft and spacecraft separation system were removed from third stage. The vibration experiment was left in place to avoid possibility of damaging third stage in removal of studs.

A complete system check of Flight Unit 1 revealed satisfactory operation.

April 16:

An RF check of the vibration experiment was conducted.

April 17:

The vibration experiment was checked and found to be operating satisfactorily.

April 18:

Spacecraft and separation system were reassembled on the third stage, and the resulting combination was placed on the balance machine.

April 19:

Balance was completed with addition of 164.5 grams in weights—less than needed in balance for first launch attempt.

April 21:

Third stage was mated with second stage.

April 23:

T-1 day checkout of spacecraft and vibration experiment were completed on T-3 day.

April 24:

Strip coating was removed from spacecraft. Spacecraft thermal coatings were touched up, and its batteries were charged.

April 25:

Thermal-coating touchup was completed.

April 26:

After completion of countdown at 1200, launch vehicle failed to fire.

Launch vehicle problem was resolved, and countdown reverted to T-15 minutes. Thereupon, countdown proceeded to lift off at 1800:1703Z.

Appendix E

Functional Test Program

Introduction

General

The functional tests were conducted on inertially correct mockups of the Ariel I spacecraft and final stage X-248 booster between October 16 and November 16, 1961. All tests were run under vacuum conditions in the 18.3m (60 ft) sphere at Langley Research Center (LRC). The operations required considerable inertia estimation, measurement, and compensation to provide correct spin-axis inertia for each test.

Equipment

The spin-drive system consisted of a vertically mounted 10 hp motor with a magnetic clutch driving an extension shaft having slip rings and a disk with 32 radially protruding studs, which generated pulses in an adjacent magnetic pickup to yield a Visicorder record of speed changes.

Two types of spin table assembly were used with the drive. For all testing involving the tall (12 + ft) Ariel I plus X-248 test item, it was necessary to couple the drive to the diaphragm of a Scout spin bearing and skirt assembly, although the frictional speed decay of this assembly was excessive. It was possible to effect some improvement in the "coasting" by using a small suspension bearing at the top of the test item. The suspension cable tension was adjusted to support most of the weight.

For the stretch yo-yo test series, where better coasting characteristics were essential, a small support bearing was used, and a 58-cm (23-in.)-diameter magnesium table was made. The spacecraft structure shells were attached to inertia disks that were attached to the magnesium table. The small [approx. 10.2 cm (4 in.) outside diam.] support bearing and 2.86-cm (1-1/8 in.)-diameter extension shaft were adequate and suitable for the compact stretch yo-yo structure models, but were structurally inadequate to safely spin up the tall top-heavy Ariel I composite assemblies. By suspending the composite assembly by an overhead cable and spinning it by hand, it appeared that substantial unbalance existed; and, even if accurate balancing could have been done, precise alignment could not have been insured.

Spinup

General

Two tests were run, using an inert-loaded X-248 furnished by LRC, with the Ariel I spacecraft shell with all appendages folded, installed on the Scout spin bearing and skirt, with drive disconnected. Inertia was estimated at 6.844 slug ft^2, comparable to flight condition.

First Spinup and Results

The first spinup test was conducted at 0.35 mm Hg. Objectives were to determine final speed and to check Lyman-alpha sensors for contamination. Results were a final speed between 120 and 130 rpm and no visible contamination of the sensors, which were returned to GSFC for further examination. The indicated final speed is not considered conclusive. The impulse rating of the PET rockets was below that required by the system, and the appearance of the PET rockets after firing indicated incomplete burning. Also the breakaway torque and running friction of the bearing was excessive (breakaway torque, 4 lb-ft +), and misalignment was suspected. The contamination check has value, but it is limited by the incomplete environmental simulation—that is, vacuum and spin environment was simulated, but zero-gravity environment was not.

Second Spinup and Results

The second spinup test was conducted at 10 mm Hg, using higher impulse PET rockets and a different skirt bearing, with part of the weight supported by an overhead bearing. Friction of the system was substantially lower. A final spin rate of approximately 180 rpm was attained. This is comparable to the flight objective.

Despin

General

These tests were performed with the spacecraft structural shell inverted on the spin table with all appendages folded up and secured to the expended X-248 by the tiedown hardware (Figure E1). The assembly was slugged up to match the flight inertia, and appendages were restrained by rubber cords adjusted to compensate for gravity effect. It became evident that the low-friction table assembly with the small support bearing would not accommodate the unbalance of this assembly. So the Scout spin bearing and skirt were used, with a suspension cable and swivel-mounted bearing adjusted (with a turnbuckle and spring scale) to take most of the weight of the assembly. The spin axis of the assembly was carefully aligned, and some effort was made to adjust the cable tension for minimum running friction. This adjustment was complicated by problems of dynamic resonance of the support cable in the operating speed range and the need to anticipate dimensional changes due to evacuation of the chamber and/or temperature changes.

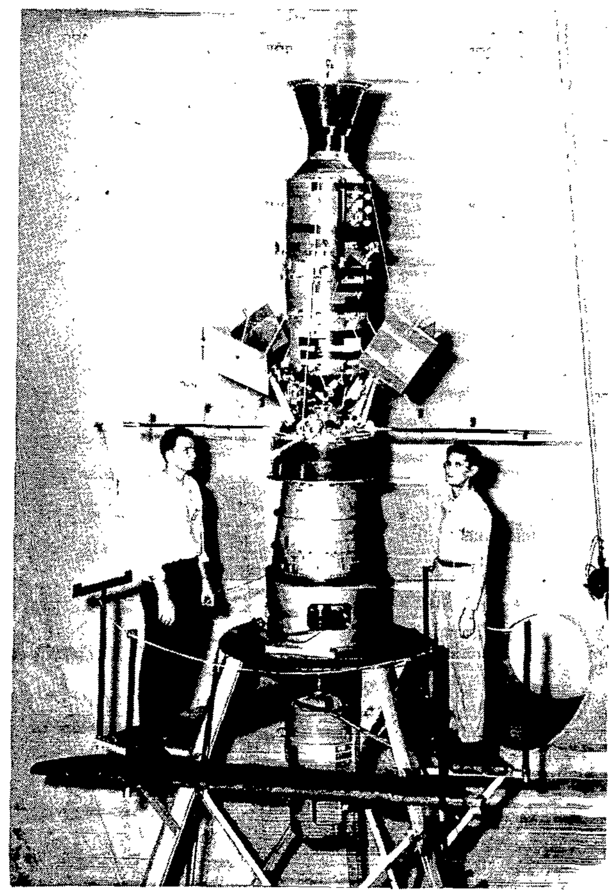

Figure E1—Setup for boom and paddle erection test.

Temperature changes had the most effect on cable tension. After the tests it was apparent that running friction was undesirably high, but marginally acceptable.

Before each operational test, it was necessary to calibrate the running speed and speed decay so as to insure planned speed at the initiation of the operation. This required reduction of recorder tape data before each step. Sequence tests were conducted at 10 mm Hg vacuum.

90 Percent Flight Spin

All operations were initiated at 90 percent of nominal flight spin rate and recorder data on speed taken before, during, and after each operation. Photo coverage at 1000 frames/sec from three angles was also obtained of each operation. Operations were triggered manually, via slip rings and hard lines to control room. As a preliminary test, the spacecraft sequence timers were caused to cycle, while spinning at nominal flight speed (160 rpm) in vacuum; but the timer contacts ignited flash bulbs mounted on the payload structure instead of actuating release mechanisms. The time sequence of the flashes was noted, and met the performance specification for the timers. Sequential operation of yo-yo despin, experimental boom erection, and solar paddle deployment was then accomplished as planned except that the experiment booms failed to lock into extended position and had to be locked in by overspinning.

110 Percent Flight Spin

This was intended to be a repeat of the first sequence test, but with operations conducted at 110 percent of nominal speed and sequence timers operated 30 rpm after despin and erection events. However, initiation of the first event (yo-yo despin) fired all release squibs (except separation squibs, which had not been installed); and yo-yo despin, experiment boom erection, and mass boom and solar paddle deployment all occurred simultaneously while spinning at 176 rpm. The reason for this misfire was discovered to be faulty wiring in the firing box—the three firing switches were inadvertently wired in parallel. The error did not show in the preceding test because firing leads were connected only as needed for each individual operation.

The results of this accidental overtest, of some value in showing structural weak points, were: speed change was violent; yo-yo operation was normal except that one wire snagged a broken paddle but snapped loose; experiment booms snapped nylon string of escapement mechanism, but booms survived extreme deflection without structural failure; both inertia booms snapped into detent, bounced back, and on rebound tore off both attachment brackets by shearing the screws—the inertia booms were restrained by the rubber cords, but one boom swung around and struck the side of the assembly; both single-hinged paddles sheared off the attachment screws at the hinge, broke the bungee cords, and fell to the floor of the chamber; both double-hinged paddles stayed on, but the detent pins in the secondary hinges were sheared off; all appendage attachment screws were loosened and elongated; the single-hinged paddles opened a little ahead of the double-hinged paddles, and the inertia booms slightly ahead of the experiment booms; all appendages opened within about 1/4 second; and the yo-yo disengaged from hook about 1/2 second after release. Final spin rate was about 30 rpm. The sequence timers were operated after the spacecraft assembly coasted to a stop. One timer functioned normally; the other failed to operate but was later found to be operable. The failure was found to be due to a leaky battery case, which is not considered significant because the overnight vacuum soak was not representative of flight conditions—neither was the extended location and operation of the batteries in an inverted position. The spacecraft assembly was reconstructed for another try—all damaged parts were replaced, and the firing box wiring was corrected.

Final Despin

This was a repeat of the first successful sequence test except that all operations were run at 110 percent of nominal speed and the sequence timer check was omitted—no replacement for the defective battery was available, and the reliability of the timers before and after exposure to despin was considered to have been established. The operational sequence was again accomplished as planned except that the experiment booms again failed to lock in extended position, despite intentional initial overspin estimated to compensate for friction-induced decay in speed during erection time of approximately 4.5 sec.

Test Results

The Visicorder records plotted speed vs. time curves for the despin operations; and, after due allowance for friction-induced speed decay, the results showed good correlation with theoretical predictions. The only significant deficiency was the failure of the experiment booms to lock in erected position.

GSFC Tests

Additional tests were run at GSFC using a test rig substantially the same as at LRC without vacuum simulation. As a result, the escapement mechanism for the experiment boom was successively modified until full deployment was achieved at minimum anticipated flight-spin rate.

Separation Tests

General

These tests were performed with the expended X-248 installed with nozzle down in the Scout skirt. The Dutchman separation device and the spacecraft shell with all appendages extended were installed on top of the X-248. The rubber cords and the batteries were removed from the spacecraft shell. The separable configuration weighed 23.1 kg (51 lb), closely approximating the requisite mass to achieve the in-flight separation velocity. A device to compensate for gravity effects during separation was installed. This consisted of an overhead bearing and support cable, with the cable wound around a conical grooved pulley and connected to a long calibrated rubber spring under initial tension. This device operated as a constant force spring, over a limited range, to counteract the weight of the separable configuration. A ratchet device was used to prevent slipback. Separation was effected by firing the release squibs of the separation device through slip rings and a hard line to the control room. Separation tests were conducted at 10 mm Hg vacuum.

Test No. 1

Separation was effected by firing both diametrically opposed release squibs, while the assembly was spinning at approximately 30 rpm (lower than the intended 39 rpm). Results were satisfactory, by visual observation and later review of slow motion movies. The spacecraft separated straight and true, with no apparent tipoff, and was smoothly arrested by the counterweight device after rising approximately 3 ft. The spacecraft continued rotating for over 13 min on the small suspension bearing, with very little wobble.

Test No. 2

This was a repeat of the first separation except that only one release squib was fixed and spin rate at separation was about 20 rpm. The spacecraft again separated with no apparent tipoff. For both separation tests the spin rates were lower than planned—perceptible slowing down was noted in the few seconds preceding separation. Visicorder data for the preceding tests indicated the same phenomenon, to a lesser degree; and it was realized that the change in voltage caused by the photoflood electrical load was responsible. The remedy was to calibrate the speed with floodlights on, and this was done for the later tests.

Results

The separation device operated satisfactorily under the test conditions—that is, 10 mm Hg vacuum, spin rates as noted, spacecraft mass adjusted for flight condition relative to separation velocity, gravity effect on the spacecraft effectively nullified, spacecraft balanced, and spin axis at separation constrained to be substantially in alignment with geometric centerline of spacecraft, X-248, and separation spring. The spin-axis constraint condition is a limitation of simulation technique, comparable to flight situation only if adequate balance of spacecraft and of expended X-248 is assumed. This is not a valid assumption, as expended boost units characteristically have some unbalance. Therefore the tests verified alignment of the separation spring thrust vector with the geometric centerline of the assembly in that no tipoff was apparent, but are inconclusive concerning possible tipoff due to dynamic unbalance at separation.

Stretch Yo-Yo System

First Series

Procedures: The Ariel I stretch yo-yo system with preloaded springs was despun from spin rates 80 and 120 percent of nominal. Designed resultant spin rate is 73.9 rpm. All tests were conducted at 10 mm Hg. Initial spin was adjusted to allow for speed decay after declutching. Despin was initiated manually, via slip rings and a hard line to the control room. Each test configuration was slugged up to the inertia of the flight system. Visicorder and photographic records were made of each run.

Results: System inertia was 2.774 slug ft^2. Results were:

1. Spun at 160 rpm. Direction of rotation was incorrect, and weights and springs fell off on release.

2. Despun from 162.6 rpm to approx. 73.6 rpm.

3. Despun from 129.4 rpm to approx. 73.8 rpm.

Second Series

Results: The stretch yo-yo system was operated nine times in vacuum conditions, and the despin effect was noted for various combinations. For nominal initial spin and inertia the final spin rate was approximately as predicted, the effect of inertia changes was as expected, and the stretch yo-yo compensated satisfactorily for initial underspin. However initial overspin, especially when combined with higher inertia, causes yielding of the yo-yo springs and overcompensation. As the overspin-correction discrepancy is attributed to yield of the spring material, it was planned to repeat pertinent tests using the higher yield point springs.

Post Test: Springs were heat-treated for higher yield points.

Third Series

Procedure: The same as for previous tests.

Results: Three tests were run, using yo-yo springs that had been heat-treated for high yield point. Springs used in earlier tests had yielded excessively. Improved performance was noted, although slight yield did occur at 25 percent over nominal spin rate. Observed results satisfy flight requirements but show slight deviation from design objectives.

The system was designed to give a final spin rate of 73.9 rpm, with nominal initial speed considered 160 rpm and nominal inertia considered 2.885 slug ft^2. The low final spin obtained in series no. 3, where slight yield occurred, is consistent with previous findings that spring yield will cause overcompensation of high initial spin. The high final spin obtained in series no. 1, followed by almost exactly the design spin rate for series no. 2, cannot be explained. Neither error is sufficient to compromise the Ariel I mission; therefore, as a flight item qualification test, the results are satisfactory. From the point of view of design theory verification it must be noted that the deviations from designed final spin, when no yield occurred, exceed the likely experimental error.

NASA-Langley, 1964 G-407